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1999 Research Engineering Annual Report

Compiled by
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NASA Dryden Flight Research Center
Edwards, California

August 2000

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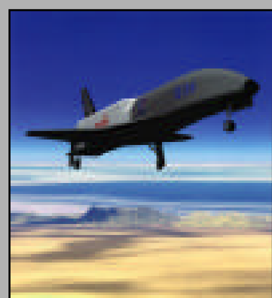

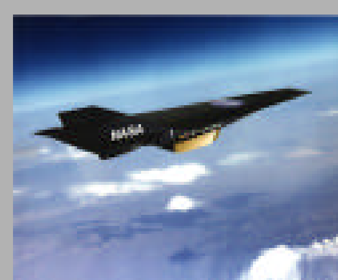

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1999 Research Engineering Report

Table of Contents

Preface

The NASA Dryden Flight Research Center's Research Engineering Directorate is a diverse and broad-based organization composed of the many disciplinary skills required to successfully conduct flight research. The directorate is comprised of six branches representing the principal disciplines of: Aerodynamics, Dynamics and Controls, Flight Systems, Flight Instrumentation, Propulsion and Performance, and Aerostructures. The Directorate organization is illustrated on the chart following this page.

We are very proud of the many significant accomplishments of our technical staff during the calendar year 1999. These milestones include both those accomplished in support of research programs as well as basic research performed within the Directorate, supported by our competitively-funded Flight Test Techniques and Disciplinary Flight Research programs.

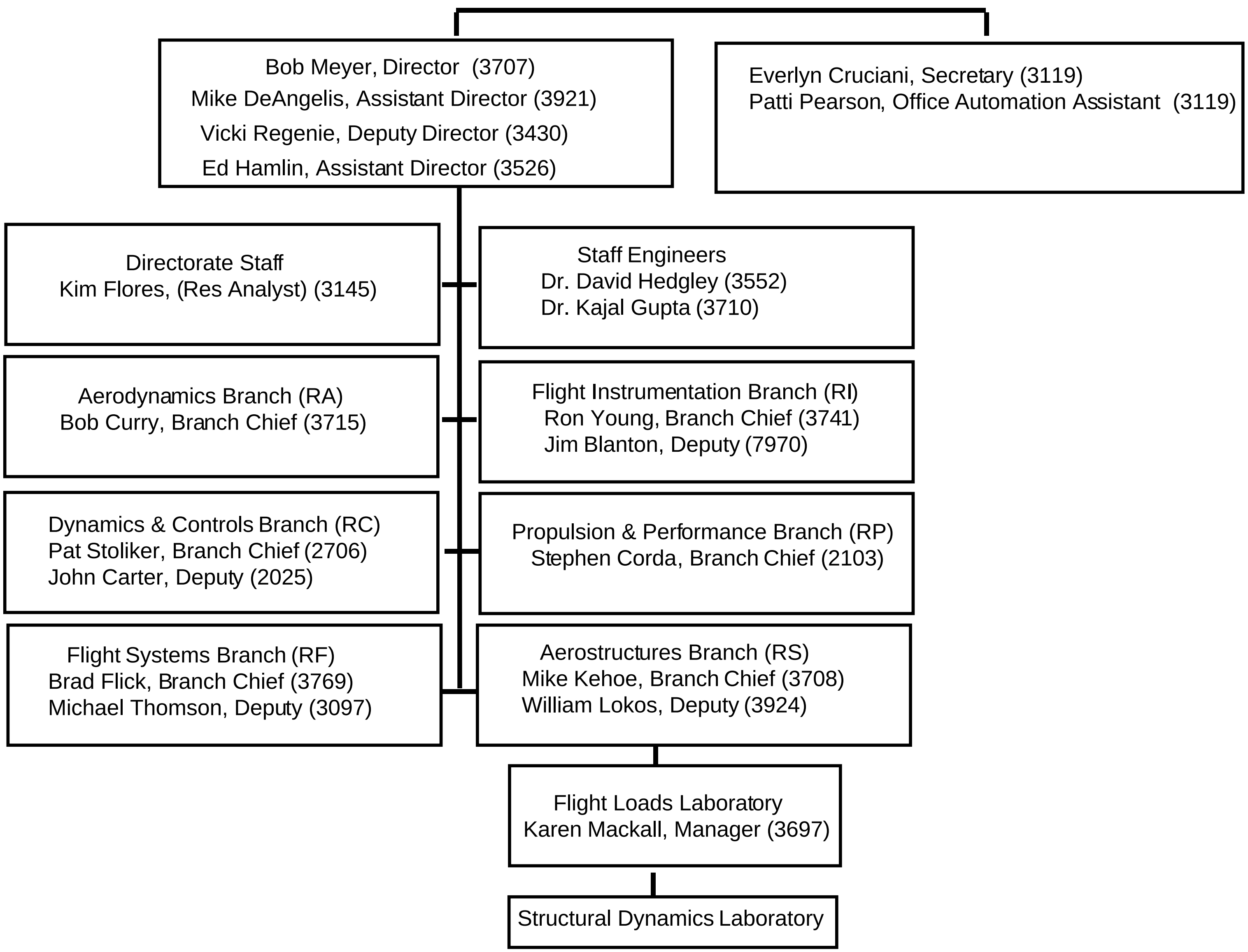
This Annual Report encompasses the full range of research accomplishments, from the project level, flight test techniques, to disciplinary flight research programs. It includes one-page summaries of each program; more details are available from the principal investigators as noted on the summaries. There are also included a list of the many technical publications completed in the last year, from in-house, university, and contract researchers under the auspices of the Directorate.

Calendar year 2000 promises to be an even more productive year, with a mix of new and continuing research programs. I look forward to reporting on these efforts next year.

RRMeyer

R.R. Meyer
Director of Research Engineering
Dryden Flight Research Center

Research Engineering Directorate (R)



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Assistant Director	Michael DeAngelis
Assistant Director	Edmund M. Hamlin
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RF – Flight Systems	Bradley C. Flick
RI – Flight Instrumentation	Ronald Young
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RS – Aerostructures	Michael W. Kehoe

Supersonic Natural Laminar Flow Flight Research

Summary

Airfoil designs capable of passively maintaining laminar flow at supersonic speeds have been shown by theory and small scale tests. Recent flight tests now prove that these designs can maintain large runs of laminar flow at higher Reynolds numbers in a harsh flight environment. Laminar flow was measured using a commercially available infrared detector adapted for flight. These tests were performed on the Dryden F-15B aircraft.

Objective

• Acquire IR images of natural laminar flow up to M=2.0.
• Obtain large runs of laminar flow at highest unit Reynolds number possible with F-15B aircraft.
• Determine conditions where laminar flow breaks down.

Approach

Using an aircraft mounted infrared camera, laminar flow was measured on a test article mounted on the centerline pylon of an F-15B aircraft. The IR camera measures surface temperatures which change with the different boundary layer states. The surface beneath the turbulent boundary layer will be warmer than that beneath the laminar layer due to the higher convection of the turbulent layer with the freestream. The laminar flow, for these conditions, will be darker and the turbulent flow lighter.

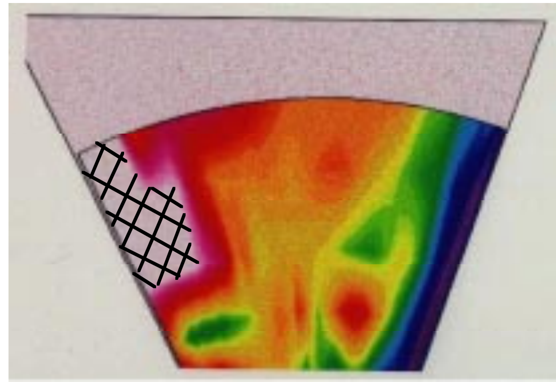
The test article was fabricated from aluminum with an insulating layer covering all but the first 1 to 2 inches of the leading and trailing edges. A splitter plate was formed over the test wing to minimize disturbances from the bottom of the aircraft effecting the test surface.

Results

Laminar flow was obtained up to full chord for the outer 1/3 span and approximately 80% of the inner 2/3. The laminar flow was able to penetrate weak shock waves, but was typically terminated by strong shock waves. The strongest shock wave appeared to emanate from the camera pod located outboard on the armament rail.

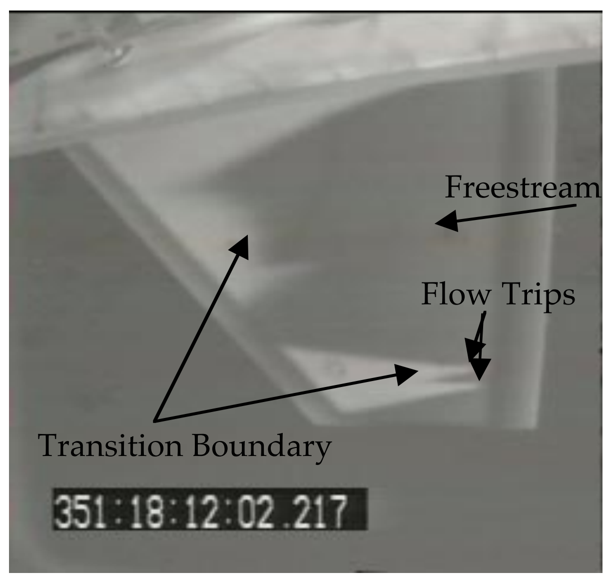
Status/Plans

The next flights will assess cross-flow disturbances by flying with a 30 degree leading edge. Future flights will obtain more detailed data by instrumenting the test article with surface pressures and thermocouples. Also the data recording system will be updated to record the full digital images from the camera. A follow on program would increase the size of the test article and incorporate potential control surfaces to assess the effects of higher Reynolds numbers and systems issues.



Predicted transition pattern of test article (cross-hatched area denotes transition).

Supersonic Natural Laminar Flow test article mounted on F-15B aircraft.



Infrared image of Supersonic Natural Laminar Flow test fixture.

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Infrared Flight Test Techniques

Summary

Infrared thermography is a very powerful technique to both visualize and quantify flow fields over the speed range. It is both global and non-intrusive. It is most widely used in aerodynamic flight test to measure laminar/turbulent transition, but can also identify shocks and measure aeroheating. Three different techniques of acquiring the IR thermograms are employed. These are a fixed system on the aircraft to be visualized, a remote system on a second aircraft, and a ground based remote system employing long range optics. These systems and techniques have been used to visualize flow fields from subsonic to hypersonic speeds.

Objectives

- Acquire infrared thermograms in flight.
- Process images for motion and spatial corrections, and enhancement.
- Calibrate and measure surface temperatures
- Identify and locate boundary layer transition, shock waves, and other flow phenomena.

Approach

By using various IR systems and techniques flow field information has been acquired across the speed range. Remotely mounted aircraft systems have been used to acquire data from other unmodified aircraft in flight from subsonic through low supersonic speeds. Fixed systems have been used to measure test articles at supersonic speeds but are applicable for a variety of applications and speed ranges. Ground based remote imaging (Langley lead for this effort) has been developed for measuring transition and aeroheating of re-entering re-useable launch vehicles.

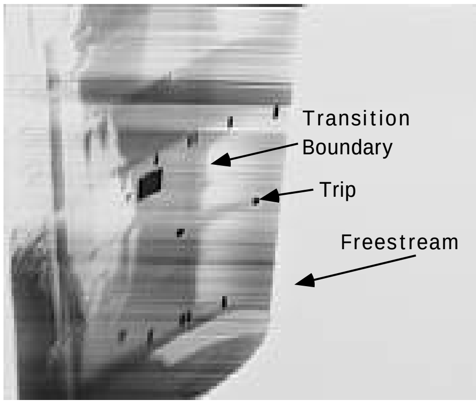
Results

Excellent results have been obtained from each type of system and at each speed range. The ability to locate boundary layer transition, shock waves, and measure surface temperatures has been demonstrated.

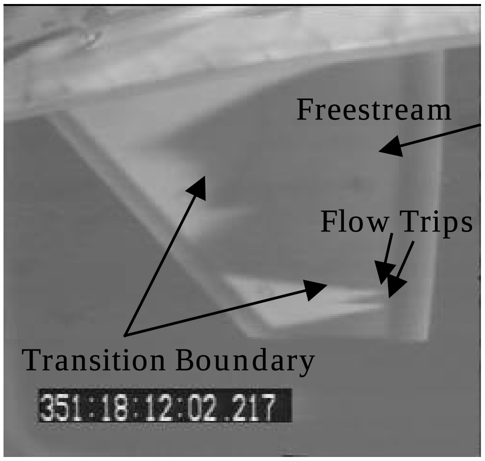
The top figure shows transition on a subsonic wing. The surface is heated by solar radiation so that the laminar regions are lighter in color. The middle figure shows transition on a supersonic laminar flow test article. In this case the surface is heated by the freestream, so the laminar areas are darker. The bottom figure shows the space shuttle on re-entry. In this case the lighter areas denote regions of higher surface heating (entire vehicle is turbulent).

Status/Plans

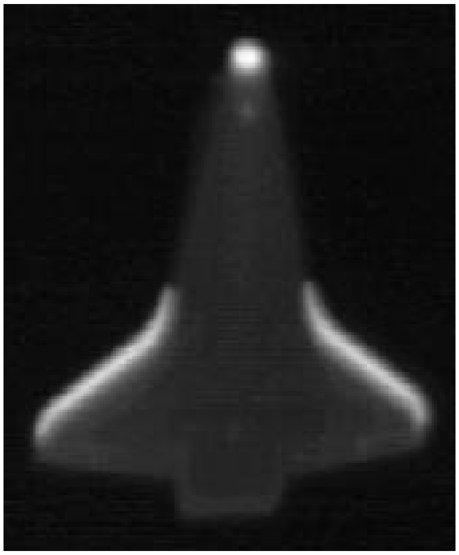
Continue developing infrared systems and techniques for flight research.



T-34 wing imaged remotely by F-18 FLIR.



Infrared image of supersonic natural laminar flow test fixture imaged with a fixed IR system.



Space shuttle STS-96 imaged with ground based IR.

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Robert C. Blanchard, LaRC

HAVE RECKON Project

Summary

When an Uninhabited Aerial Vehicle, UAV, is piloted from a ground control station, it is referred to as a Remotely Piloted Vehicle. A delay of typically 0.1 seconds is incurred as the vehicle processes the uplink signal and the vehicle response is prepared for downlink. When an RPV is flown over the horizon, the uplink and downlink signals still need to be relayed back to the ground station. A desirable relay is to use a satellite as the link. However a satellite link to the UAV and back often results in an additional signal transport delay of about 0.4 seconds. This 0.5 seconds total delay is enough to significantly degrade the handling qualities of an RPV. For a high-gain piloting task, this may even be enough lag to make the vehicle uncontrollable.

The USAF Test Pilot School, TPS, proposed to enhance the handling qualities by using a model-based predictive algorithm to compensate for system time delay. The algorithm would reside within the ground station and provide the ground pilot with predictive enhancements overlaid on the ground monitor. The compensation was functionally independent of the UAV. To conduct a low-cost flight research experiment, TPS contacted the Dryden Subscale Facility (model shop). Dryden agreed to provide the "UAV", safety pilot, ground cockpit, telemetry van and other lesser assets for the flight tests. The scope of the experiment was limited to the longitudinal axis.

Objective

The USAF TPS general objective was to determine the improvement in UAV handling qualities when using a model-based predictive algorithm to compensate for system time delay. NASA Dryden's primary objectives were to provide the TPS with an aerodynamic model of the flight vehicle, integrate the vehicle systems, and safely conduct the flight tests.

Results

Dryden chose a large utilitarian model, called the Mothership, that was capable of lifting 25 lbs of payload and still remain within the Dryden definition of a model airplane. A 16-channel data system was carried in a pod under the main fuselage. An airdata noseboom was mounted on the nose. A forward-looking video camera was installed in a pod above the wing. Measurements of elevator, pitch rate, and angle of attack were downlinked on the audio channel of the video signal. A telemetry van was used to separate the data from the audio channel and put it onto an ethernet line to an USAF Unix workstation next to the research cockpit. The workstation generated the transport lag by passing a delayed video frame to the monitor. The workstation was also used to generate the predictive algorithm and the related monitor display. The research cockpit and video monitor was in a separate van. Altitude and airspeed were merged with the video signal onboard the model and overlaid onto the upper corners of the video image.

An initial longitudinal aerodynamic model of the Mothership was calculated using vortex-lattice techniques. Parameter estimation analysis of elevator doublet flight data was then used to update the update the initial model. A total of 28 flights were flown for the initial functional checks and for parameter estimation.

The NASA model pilot, referred to as the "safety" pilot, stood outside of the van containing the ground cockpit. He was pilot in command and insured that the vehicle was kept within his visual range. He made all the takeoffs, landings, climbs, descent, and turns. On the straight leg of the flight pattern, he would give longitudinal stick and throttle control to the research pilot within the van. He could take back control at any time.

For the research flying of the program, 62 data sorties were flown over 16 test days for 20.5 hours of flight time. TPS was able to show that their model-based predictive algorithm improved handling qualities of the model flown as an UAV. With the predictor, the time delay tolerated was significantly increased. Or at the same time delay, pilot workload was significantly decreased. Including the initial checkout flights, a grand total of 90 flights were flown.

Status/Plans

Dryden started in May, 1999. The research flight testing started in mid September and ended in mid October, 1999. USAF TPS technical report was finalized in December 1999. The successful completion of the program has potential to lead to a joint USAF/NASA program using a much larger vehicle.

Mothership Utility Model with 10 ft Wing Span

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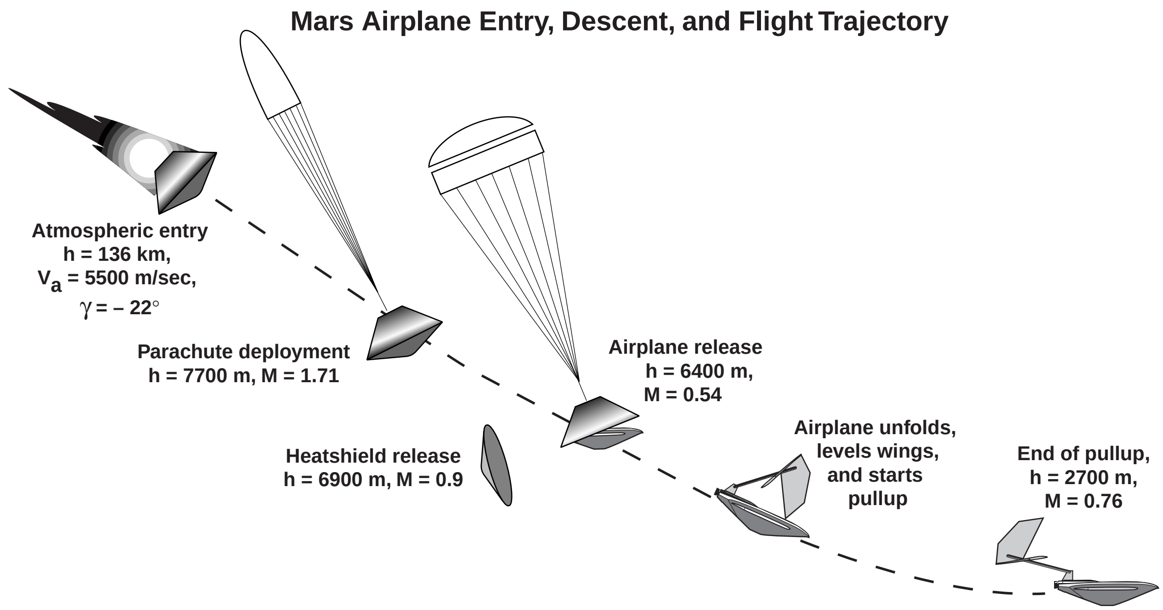
Mars Airplane Entry, Descent and Flight Trajectory Development



Summary

The Mars Airplane project was to provide the first opportunity for sustained lifting atmospheric flight on an interplanetary mission. For the mission, the aircraft is carried to the planet folded inside a small (75 cm diameter) aeroshell which makes a direct entry into the Martian atmosphere. Critical to the mission was the transition from the hypersonic, ballistic aeroshell to the subsonic lifting airplane. For initial design purposes a baseline Entry, Descent and Flight (EDF) trajectory profile was defined to consist of the following elements: ballistic entry of the aeroshell, supersonic deployment of a decelerator parachute, release of the heatshield, release, unfolding, and orientation of the airplane, and execution of a pullup maneuver to achieve trimmed, horizontal flight.

Given the predicted mass and aerodynamics of the aeroshell and airplane, and the predicted Martian atmospheric and topographical characteristics, a simulation of the EDF trajectory was built using the Program to Optimize Simulated Trajectories (POST). The trajectory optimization feature of POST was used to find the control strategy which maximized the surface-relative altitude of the airplane at the end of the pullup maneuver, subject to a number of design constraints.

Objectives

The early development of a baseline (i. e. first-generation) EDF trajectory was important to provide the Mars Airplane design team with essential feedback early in the design cycle. The initial objective was to develop an EDF trajectory consistent with the baseline vehicle design data, and to optimize the trajectory within the allowable design space. With the optimal trajectory defined, the objective was to identify the sensitivity of the trajectory to design parameter variations. A final objective was to use the results of sensitivity analysis to develop a second-generation control strategy.

Results

For the baseline vehicle design, viable EDF trajectories were developed for sites in both the northern and southern hemispheres; altitudes at the end of the pullup maneuver were above 2 km. However, the high elevation of the desired science mission site (Parana Valles) precluded completion of the pullup maneuver before surface impact.

The performance metric (the surface-relative altitude at the end of the pullup maneuver) was a most sensitive to airplane mass, airplane lift and drag coefficients, and maximum Mach number allowed during the pullup. Not surprisingly, decreasing airplane mass, increasing lift coefficient, and increasing the allowable maximum Mach number yielded net altitude gains. Increasing the airplane drag coefficient to yield a net altitude increment. The performance metric showed only small sensitivity to other design parameter variations studied.

With the addition of airplane drag as a design variable, a second-generation control strategy showed that increasing drag significantly during the pullup maneuver yielded a net altitude increment of about 1.7 km.

Status/Plans

The Mars Airplane project was cancelled near the end of CY99. However, documentation of current results and some closeout work continues.

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Blunt Body Drag Reduction Using Forebody Surface Roughness

Introduction

Current proposed shapes for trans-atmospheric launch and crew-return vehicles like the X-33, X-34, X-38, and "Venture-Star" have extremely large base areas when compared to previous hypersonic vehicle designs. As a result, base drag - especially in the transonic flight regime -- is expected be very large. The unique configuration of the X-33, with its very large base area and relatively low forebody drag, offers the potential for a very high payoff in overall performance if the base drag can be reduced significantly.

Drag Reduction Strategy

For blunt-based objects whose base areas are heavily separated, a clear relationship between the base drag and the "viscous" forebody drag has been demonstrated (ref. 1, 2). (Figure 1) As the forebody drag is increased; generally the base drag of the projectile tends to decrease. This base-drag reduction is a result of boundary layer effects at the base of the vehicle. The shear layer caused by the free-stream flow rubbing against the dead, separated air in the base region acts as a jet pump and serves to reduce the pressure coefficient in the base areas. The surface boundary layer acts as an "insulator" between the external flow and the dead air at the base. As the forebody drag is increased, the boundary layer thickness at the aft end of the forebody increases, -- reducing the effectiveness of the pumping mechanism -- and the base drag is reduced.

Configurations with large base drag coefficients, necessarily lie on the steep portion of the curve a small increment in the forebody friction drag should result in a relatively large decrease in the base drag. Conceptually, if the added increment in forebody skin drag is optimized with respect to the base drag reduction, then it may be possible to reduce the *overall drag of the configuration*.

The LASRE Drag Reduction Experiment

The LASRE experiment (ref. 3) was a flight test of a roughly 20% half-span model of an X-33 forebody with a single aerospike rocket engine at the rear. The entire test model is mounted on top of a SR-71 aircraft. In order to measure performance of the Linear Aerospike engine under a variety of flight conditions, the model was mounted to the SR-71 with a pylon which was instrumented with 8 load-cells oriented to allow a six-degree-of-freedom measurement of the total forces and moments. The model was also instrumented with surface pressure ports that allowed the model profile drag to be measured by numerically integrating the surface pressure distributions.

The LASRE drag reduction experiment sought to increase the forebody skin friction and modify the boundary layer at the back end of the LASRE model. Clearly, one of the most convenient methods of increasing the forebody skin drag is to add roughness to the surface. LASRE results verified that as the forebody drag is increased; generally the base drag of the projectile tends to decrease. LASRE tests verified that this drag reduction persists through transonic and well into the supersonic flight regime. Pre-flight analyses predicted that a trade off of viscous forebody drag to base drag may make it possible to achieve a net drag reduction by adding roughness to the forebody skin.

Unfortunately, even though the LASRE experiments demonstrated a strong base drag reduction effect; the net drag was not reduced. The roughened forebody caused the forebody pressures to rise and offset the base drag benefits, which were gained. At this time it is unclear whether this forebody pressure rise was a fault of the manner in which the forebody grit was applied, or whether it is even possible to achieve a net drag reduction. Further tests, under a more controlled flow environment, must be conducted to resolve this uncertainty.

Blunt Body Drag Reduction Using Forebody Surface Roughness

Wind Tunnel Tests A series of low-speed wind tunnel tests is currently being conducted to prove the existence of this elusive Drag Bucket. In these tests a two-dimensional cylinder with a blunt after body is being tested (Figure 2). For the simple 2-D tests total body force measurements will be determined by numerically integrating surface pressures and by skin drag calculations made using the boundary layer velocity profile measurements. High-frequency wake pressure (Strouhal number) measurements will also be obtained. By adding micro-machined surface overlays (Figure 3) to increase the surface roughness it is hoped that the overall drag of the configurations can be reduced. The predicted drag reduction is shown in figure 4. The body flow characteristics predicted by CFD analyses are shown in figure 5. The trailing Von-Karman “vortex-street” wake is clearly visible. Significant instrumentation system development was achieved during FY ‘99. Test results confirming these predictions should be available by mid year FY ‘00.

Figure 1: Visualization of the "Drag Bucket"

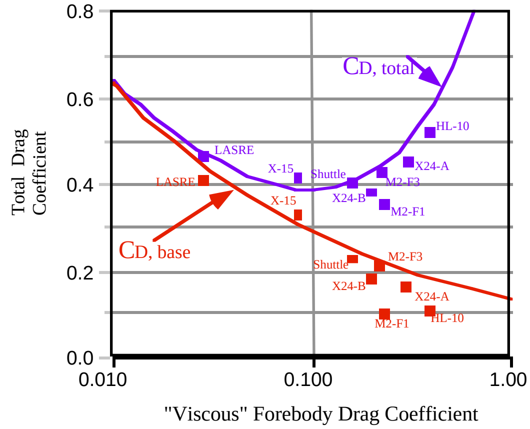


Figure 2: Wind Tunnel Model for Base Drag / Forebody-Roughness Study:

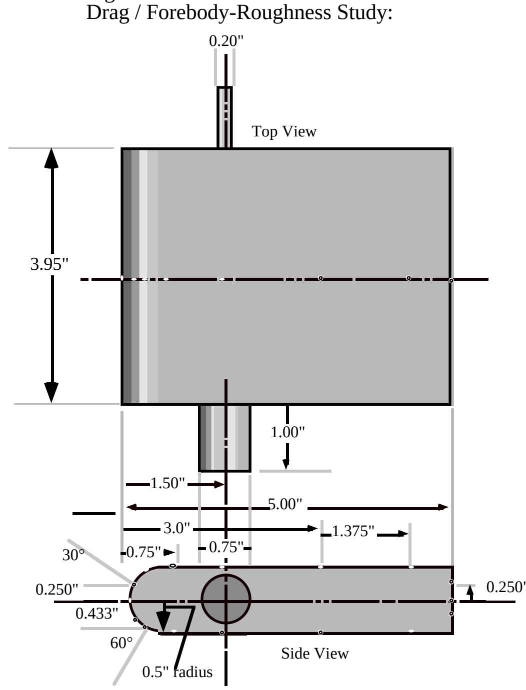
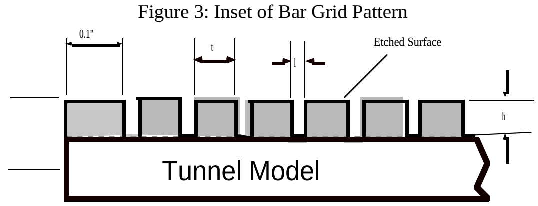


Figure 3: Inset of Bar Grid Pattern



References

1) Hoerner, Sighard F., *Fluid Dynamic Drag*, Self-Published Work, Library of Congress Card Number 64-19666, Washington, D.C., 1965, pp. 3-19, 3-20,15-4, 16-5.

2) Saltzman, Edwin J., Wang, Charles K., and Iliff, Kenneth W., *Flight Determined Subsonic Lift and Drag Characteristics of Seven Blunt-Based Lifting-Body and Wing-Body Reentry Vehicle Configurations*, AIAA Paper # 99-0383, 1999.

1) Whitmore, Stephen A., and Moes, Timothy R., *A Base Drag Reduction Methods on the X-33 Linear Aerospike SR-71 Experiment (LASRE) Flight Program*, AIAA 99-0277, January 1999.

Blunt Body Drag Reduction Using Forebody Surface Roughness

Table I: "Bar-Grid" Roughness Dimensions

(See Inset Diagram) (κ_s ~ Equivalent Sand Grain Roughness of Plate

0.0020	0.0020	0.0020	0.0064	± .0005	
0.0020	0.0040	0.0020	0.0114	± .0005	
0.0020	0.0060	0.0050	0.0206	± .0005	
0.0050	0.0120	0.0050	0.0330	± .0010	
0.0100	0.0150	0.0100	0.0450	± .0010	
0.0100	0.0200	0.0100	0.0570	± .0010	
0.0200	0.0400	0.0200	0.1140	± .0015	
0.0200	0.0800	0.0200	0.1911	± .0015	
0.0400	0.1000	0.0400	0.2721	± .0020	
0.0400	0.1500	0.0400	0.3660	± .0020	

Figure 4: Predicted Drag Reduction for Wind Tunnel Model

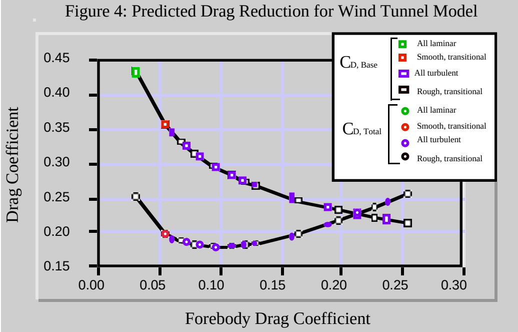
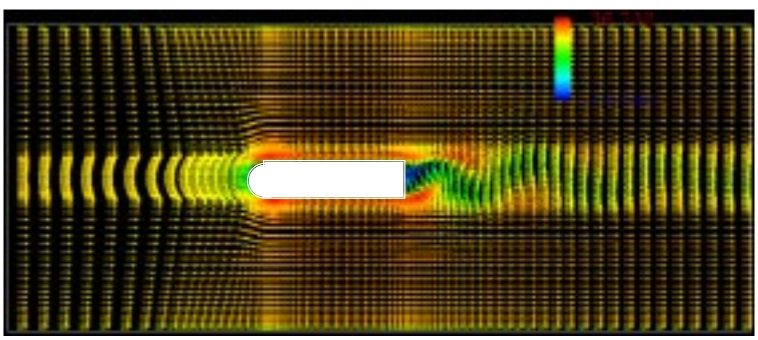


Figure 5: CFD Results for Wind Tunnel Model Showing Von-Karman



DC-8 Reduced Vertical Separation Minimum (RVSM) Certification

Summary

The NASA Dryden DC-8 Airborne Science Laboratory (N817NA) performs research around the globe, recently in support of the SAGE III Ozone Loss and Validation Experiment (SOLVE). This experiment utilized a large region of airspace, the North Atlantic (NAT) airspace corridor, which is subject to Reduced Vertical Separation Minimum (RVSM) requirements. These requirements allow aircraft traffic to be separated by 1,000 feet vertically between 29,000 and 41,000 feet MSL, as compared to the usual 2,000 feet separation. RVSM non-group aircraft compliance requires ±160 feet pressure altitude accuracy. RVSM allows greater traffic density while maintaining safe aircraft separation.

A commercial service for RVSM certification was considered, but involved significant modification to the aircraft, high cost, and an unacceptable schedule impact to Airborne Science research commitments. The approach taken was done internally with insignificant aircraft modification, minimal cost, and little schedule impact.

Objective

Obtain RVSM certification through an airdata calibration of the DC-8 static pressure system achieving ±160 feet accuracy.

Approach

RVSM quality airdata computers were installed in the aircraft, and these data were recorded using the DC-8's Data Acquisition and Distribution System (DADS). These airdata computers have a worst case avionics error of 85 feet after 24 months. For the calibration flights a carrier-phase differential global positioning system (DGPS) receiver and antenna was employed. The DGPS gave geometric altitude of the aircraft to an accuracy better than 2 feet. A pressure calibration of the atmosphere on flight test days was determined by data from a network of rawinsonde weather balloons, synoptic analysis, and surface observations.

By combining DGPS geometric altitude with the pressure calibration of the atmosphere, the true pressure altitude of the aircraft is determined. This is compared to the airdata computer measurement with no error corrections to determine the static source error correction (SSEC) required to null the pressure altitude errors. The SSEC for both the pilot and co-pilot systems were then incorporated into the airdata computers, and checked on a verification flight. The SSEC is a function only of Mach number.

The DC-8 was flown near maximum speed (Mach 0.48 to 0.54) at 500 feet above Rogers Dry Lakebed in steady flight. These data gave the SSEC with minimal uncertainty of the atmospheric pressure calibration. Most of the flight data was taken at 29,000 to 41,000 feet in stabilized flight between Mach 0.51 and 0.86. The near-ground data were used to adjust the high altitude data for small temperature biases on the rawinsonde balloons. DGPS data taken during constant airspeed turns were used to measure winds independent of the rawinsonde balloons.

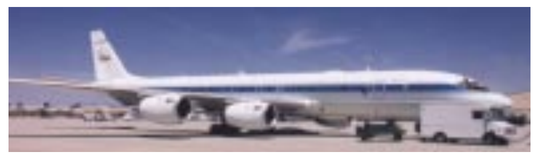
Autopilot operation was verified during stabilized flight to conform to the ±65 feet RVSM requirement.

Results

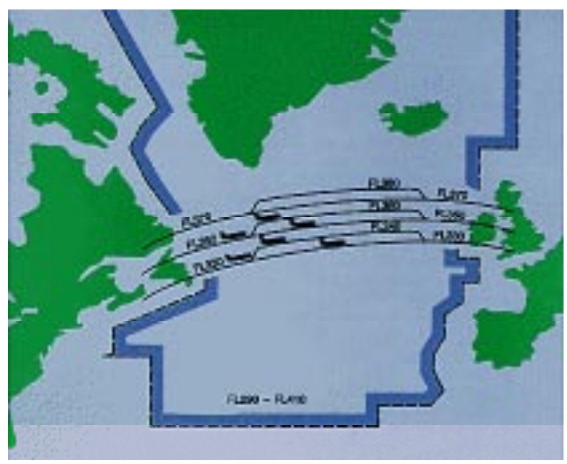
One calibration flight was flown to determine the SSEC required to null pressure altitude errors with the aircraft in a clean configuration. These data yielded error residuals of ±15 feet for both the pilot and co-pilot static pressure system.

A verification flight was performed with a variety of airborne science probes external to the aircraft, including a large nacelle about 5 feet from the static pressure ports. This constituted a worst case scenario of possible SSEC shifts. The maximum residual error on this flight was 73 feet, and when combined with the worst case avionics error of 85 feet and the DGPS accuracy of 2 feet gives a total error of 160 feet, just meeting the RVSM requirement. The errors would be considerably less if the airborne science probe near the static ports is removed or relocated. Autopilot operation was demonstrated to be within ±30 feet.

RVSM certification was granted on November 18, 1999.



DC-8 Airborne Science Laboratory (N817NA)



Current RVSM airspace in North Atlantic

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F-18 Stability and Control Derivative Estimation for Active Aeroelastic Wing Risk Reduction

Summary

The NASA Dryden F-18 System Research Aircraft (SRA) was used to obtain stability and control derivatives from flight data for a baseline F-18 aircraft. This work was done at the higher dynamic pressure range of the F-18 envelope in support of a future F-18 program known as Active Aeroelastic Wing (AAW). The AAW technology integrates vehicle aerodynamics, active controls, and structural aeroelastic behavior to maximize vehicle performance. In particular, the goal of the AAW project will be to maximize the contribution of a reduced-stiffness F-18A wing to roll rate performance. In order to support the technology, changes to flight control computers and software will be required and therefore a good understanding of the basic F-18 individual control surface effectiveness is essential The SRA was used to provide that understanding.

Objective

The primary objective of this program was to obtain detailed understanding of the effectiveness of each control surface at the various test conditions. The results can then be used to update the aerodynamic model for the F-18 and therefore improve the effectiveness of the control laws being developed for the AAW aircraft.

Approach

An on-board excitation system (OBES) was used to provide uncorrelated single surface input (SSI) doublet sequences. Longitudinal maneuvers included leading edge flap (LEF), trailing edge flap (TEF), symmetric aileron, and symmetric horizontal tail SSIs. Lateral-directional maneuvers include rudder, differential LEF, differential TEF, aileron, and differential tail SSIs. The pilot would initiate the maneuver sequence from the cockpit and the OBES would command the SSI doublets. In some cases, the surfaces were moved in combinations which are not used by the basic F-18 control laws. These included symmetric LEF, TEF, and aileron deflections at high speeds. Data was analyzed post-flight using an output-error parameter estimation algorithm.

Results

A complete set of stability and control derivatives has been obtained for 20 different Mach/Altitude test conditions. The effect of symmetric aileron deflection on pitching moment in shown in figure 1. As can be seen, the flight determined results (symbols) show that there is more pitching moment effectiveness than predicted by the simulation (especially at subsonic Mach numbers). Control surface rolling moment " reversal" was of special interest to the project. Figures 2 and 3 show TEF and aileron rolling moment derivatives. As seen in figure 2, TEF reversal was measured in flight at Mach 0.95 for altitudes below 10,000 ft. Aileron reversal was not measured in flight (figure 3), but reduced aileron effectiveness was clearly seen as the subsonic Mach number increased and altitude decreased.

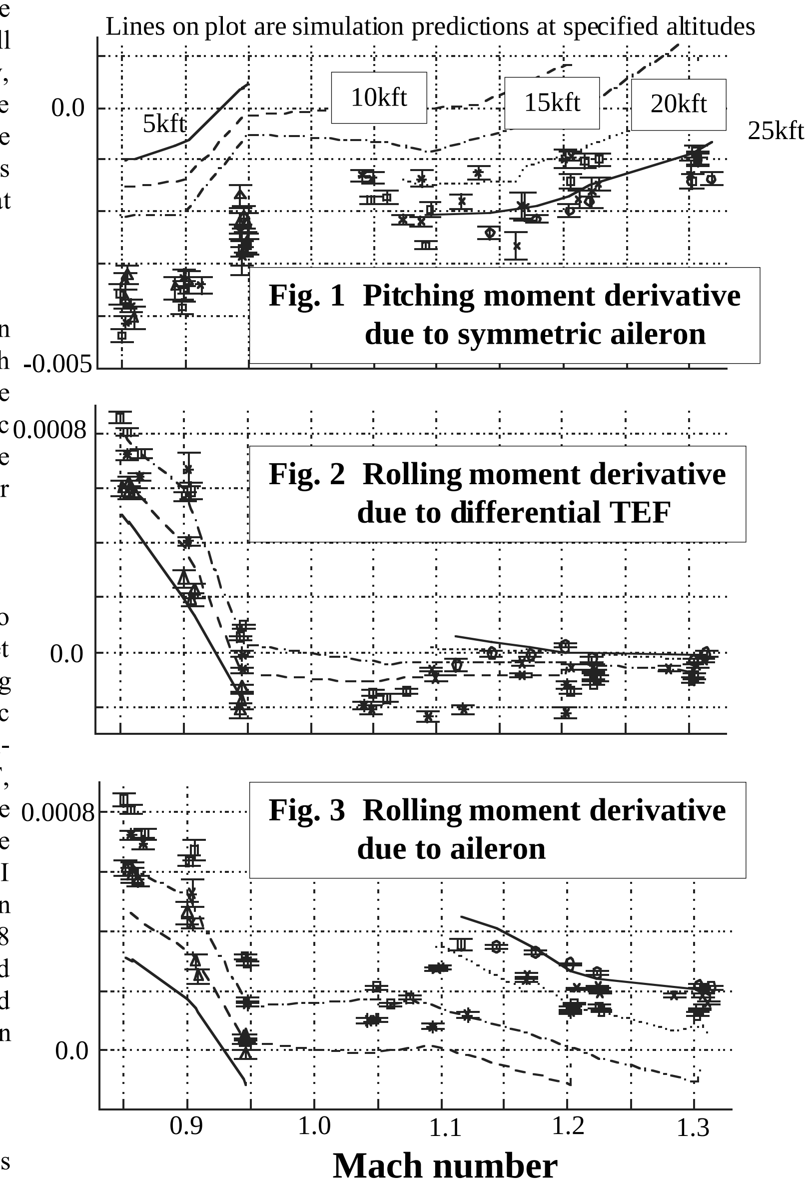

Fig. 1 Pitching moment derivative due to symmetric aileron

Fig. 2 Rolling moment derivative due to differential TEF

Fig. 3 Rolling moment derivative due to aileron

Status

The results of this study are being used to update the F-18 aerodynamic model for simulation use AAW control law development.

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SR-71 Testbed Configuration Envelope Expansion

Summary

A four flight envelope expansion program was conducted on NASA Dryden's SR-71A aircraft in the "Testbed" configuration. This configuration included a large "canoe-like" structure and reflection plane mounted on top of the SR-71 as seen in figure 1. This structure was previously used to carry the Linear Aerospike SR-71 Experiment (LASRE). Widespread interest has been expressed in using the testbed configuration for flight test of various propulsion and aerodynamic experiments at Mach numbers of up to 3.2. Consequently, NASA conducted the envelope expansion test program to look at the configuration performance, thermal environment, and stability and control (S&C).

Objective

Demonstrate maximum performance and stability and control characteristics of the SR-71 testbed configuration.

Approach

The approach was to incrementally build up Mach number on each flight. Longitudinal and lateral-directional doublets were flown at various Mach/Altitude conditions to obtain the S&C data. A post-flight output-error parameter estimation algorithm was used to obtain S&C derivatives.

Results

Performance:

The added aerodynamic drag of an experiment is most critical in the transonic acceleration, where excess thrust is at a minimum. Poor transonic acceleration can affect the maximum Mach number attainable since less fuel is available for the acceleration. On "hot" days, i.e. the temperature versus altitude profile is well above the Standard Day profile, the SR-71 J58 engine thrust is significantly reduced, thereby significantly reducing the maximum Mach capability. The SR-71 Test Bed configuration successfully reached Mach 3.0 on a "hot" day. Analysis has shown that Mach 3.2 is attainable, however, flight demonstration of this was not accomplished due to an in-flight system failure. The previous LASRE program reached a maximum Mach of 1.8 in flight. Analysis showed that the LASRE configuration could have achieved Mach 2.5 for a standard day temperature profile. The additional drag due to the part of the LASRE experiment located above the reflection plane is shown in figure 2. Future experiments desiring Mach numbers greater than 2.5 need to have drag increments less than that shown in figure 2.

Stability and Control

The largest S&C concern for the testbed configuration was reduced direction stability, Cnb. Figure 3 shows the flight determined Cnb values for the baseline and testbed SR-71 configuration. At Mach 2.1 the testbed configuration showed a steep decline in open-loop directional stability. Cnb did, however, remain positive up through Mach 3.0. The closed-loop directional stability provided by the stability augmentation system (SAS) was significantly better due to side acceleration feedback. Piloted simulations of engine unstart events were done that showed that the configuration was safe to fly with the reduced stability.

Fig. 1 - SR-71 Testbed configuration.

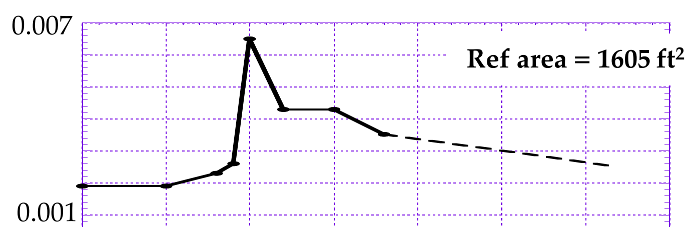

Fig.2 - Drag coefficient due to LASRE experiment

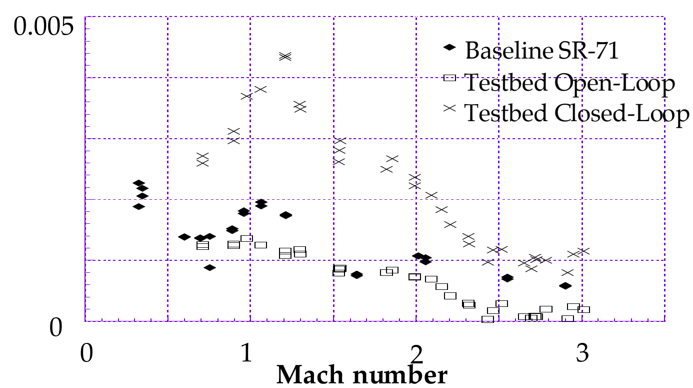

Fig. 3 - Directional Stability Coefficient

Status

The SR-71 testbed is capable for flight testing new aerodynamic and propulsion experiments at Mach numbers up to 3.2

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Flight Test of a Gripen Ministick Controller in an F/A-18 Aircraft

Summary

In March of 1999 five pilots conducted a handling qualities evaluation in an F/A-18 research aircraft of the small displacement center stick controller developed for by Sweden for the JAS-39 Gripen. The Production Support Flight Control Computers (PSFCC) provided the interface between the controller hardware installed in the aft cockpit and the standard flight control laws for the F/A-18 aircraft.

Objective

The primary objective of the flight research program was to assess any changes in the handling qualities of the F/A-18 aircraft as a result of the mechanical characteristics of the ministick. The secondary objective was a demonstration of the capability of the PSFCC to support flight test experiments.

Approach

The ministick controller and demodulator box output single channel pitch and roll stick commands. These DC signals were input into the PSFCC analog inputs. Software was developed to perform cross-channel data links, signal selection and command scaling. The signals were scaled to be similar to the maximum inputs of the standard F/A-18 control stick. Because the experiment was to assess the effects of the mechanical characteristics of a small displacement center mounted control stick, the original software deadbands and stick shaping were used.

A five flight test program was conducted using five pilots. General comments and handling qualities ratings were collected. The flight testing consisted of the following maneuvers: doublets, frequency sweeps, bank attitude captures, pitch attitude captures, echelon formation flight, column formation flight, gross acquisition, and fine tracking. There were three phases to the echelon formation: gentle maneuvering, vertical captures and more aggressive maneuvering. The three phases for column formation flight were: gentle maneuvering, more aggressive maneuvering, and lateral captures.

Results

Cooper Harper ratings from the pilots are summarized in the table below. The pilot comments consistently noted that the stick was very sensitive in roll with some tendency to ratcheting. This could be mitigated by modifying the stick shaping and deadband. The general pilots comments on the stick were favorable. It was noted that it was extremely easy to put in full amplitude inputs.

Status

Data analysis is being completed and a technical reports are being prepared to document the flight test, compare results with handling qualities criteria, and to describe the PSFCC implementation and testing process.

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Ministick installed in the aft cockpit of an F-18

Pilot	A	B	C	D	E
Echelon Formation Phase 1	4	3	2	3	3
Echelon Formation Phase 2	4	5	3	2	4
Echelon Formation Phase 3	4 to 7	5	4	4	5
Column Formation Phase 1	4	3	2	4	4
Column Formation Phase 2	3	3	2	4	5
Column Formation Phase 3	3 to 4	5	5	4	4
Gross Acquisition	n/a	6 to 7	2	2	4
Longitidinal Fine Tracking	n/a	3	2	2	3
Lateral Fine Tracking	n/a	4	2	3	6

Summary of pilot ratings

Autopilot Design for Autonomous Recovery of the APEX Remotely Piloted Vehicle (RPV)

Summary

The APEX lost-link autopilot is used when uplinked pilot command signals to the RPV flight controls are lost. This results in the APEX RPV going into autonomous flight controlled by the lost-link autopilot. During this lost-link event, the autopilot automatically returns the RPV to a safe landing or until the pilot command uplink signals are safely reestablished.

Objective

Develop and demonstrate a reliable lost-link instrument arrival, approach, and landing procedure that can be flown autonomously by the lost-link autopilot or by the test pilot from anywhere on the flight test range, and which conservatively remains within the RPV flight envelope maneuver capabilities.

Justification

Range safety requires RPVs to remain within the flight test range confined to RPV airspace boundaries either in remotely piloted or autonomous flight mode or risk flight termination. The lost-link autopilot would satisfy range safety airspace requirements by flying a conservative instrument flight procedure that remains well within those airspace boundaries during any period when uplink pilot flight command signals are lost.

Approach

The lost-link autopilot is programmed to perform a power-off, nonprecision global positioning system(GPS) arrival, approach, and landing procedure. This instrument approach procedure is also flown by the test pilot as a conservative set of maneuvers to bring the RPV back after the flight test. If lost-link occurs at anytime during the approach, the lost-link autopilot automatically completes the instrument approach followed by a safe landing. The stand-alone GPS instrument approach procedure is designed to Federal Aviation Administration specifications resulting in the required RPV approach maneuvers being well within flight envelope capabilities.

Results

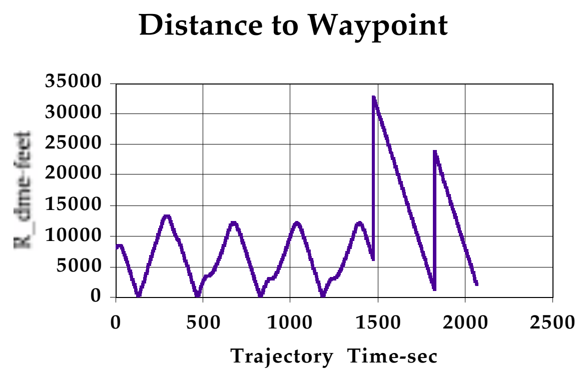
In the illustrated time history, the lost-link autopilot is executing the lost-link arrival, approach, and landing procedures. This set of maneuvers starts with the RPV arriving at the initial approach fix, followed by an entry to a holding pattern with the inbound leg aligned with the final approach to the runway used for landing. The unpowered APEX RPV remains in holding until the approach decision altitude is reached, and then executes a final approach to the runway and lands. The lost-link autopilot can successfully execute this GPS instrument approach procedure from any where on the flight test range, and remain within the RPV flight envelope.

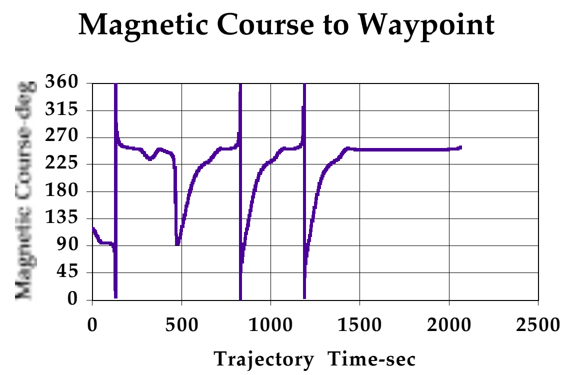
Benefits

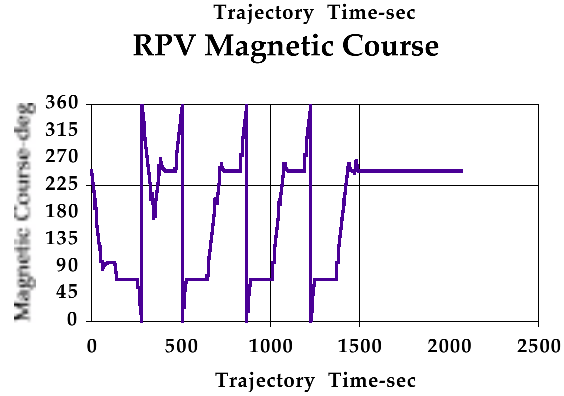
- Reduced pilot work load during arrival and approach procedures to landing because the lost-link autopilot is available to perform the same instrument approach procedures.
- Satisfies range safety requirements to keep the RPV within defined flight test airspace boundaries.
- Reduced RPV flight test risk by reliably bringing the RPV back to a safe landing during lost-link.

Status

The APEX lost-link autopilot will be reconfigured to be used on the blended wing body flight test program.







Time history of lost-link autopilot arrival,approach and landing

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AAW Risk Reduction flights using the PSFCC Computers

Summary

The Production Support Flight Control Computers (PSFCC) were used to place aerodynamic parameter identification inputs on individual surfaces for a standard F/A-18. These inputs were executed at high dynamic pressure conditions to refine the aerodynamic and loads models used for the Active Aeroelastic Wing (AAW) program.

Objective

The objective of this effort was to enhance the quality of the F/A-18 aerodynamic and loads models for use on the F/A-18 Active Aeroelastic Wing program.

Justification

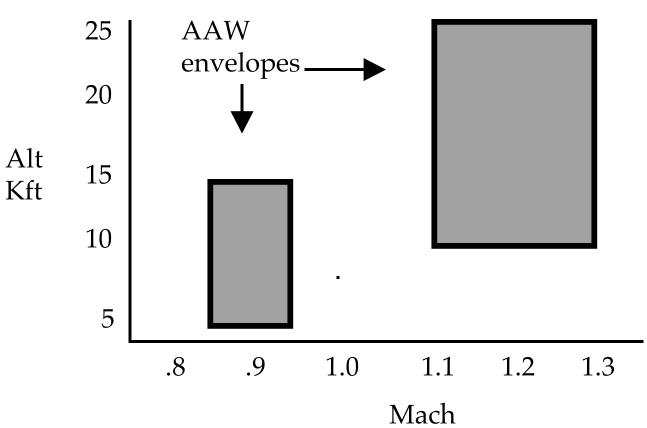
The Active Aeroelastic Wing program will place a more flexible wing on an F/A-18 to demonstrate increased performance using wing flexibility. The load and aerodynamic models used for this program are based on the best current F/A-18 models with modifications for the increased flexibility. Data from this flight program is being used to verify the F/A-18 models within the Active Aeroelastic Wing test envelopes (see below).

Approach

The research processor in the PSFCC is programmed to add doublets and Schreoder frequency sweeps to individual surface commands at pilot request (right). These doublets and surface commands were executed at 20 flight conditions within 2 regions which will be used by the Active Aeroelastic Wing program. The test vehicle had standard instrumentation for rates, accelerations, surface positions, and airdata. In addition, loads instrumentation was installed to measure various loads and hinge moments.

Results

The flight test results showed significant differences between the existing F/A-18 models for loads and aerodynamics and the flight derived values. The predictive models for the F/A-18 are currently being modified to reflect the flight test performance of the aircraft.



AAW Test Envelopes

Benefits

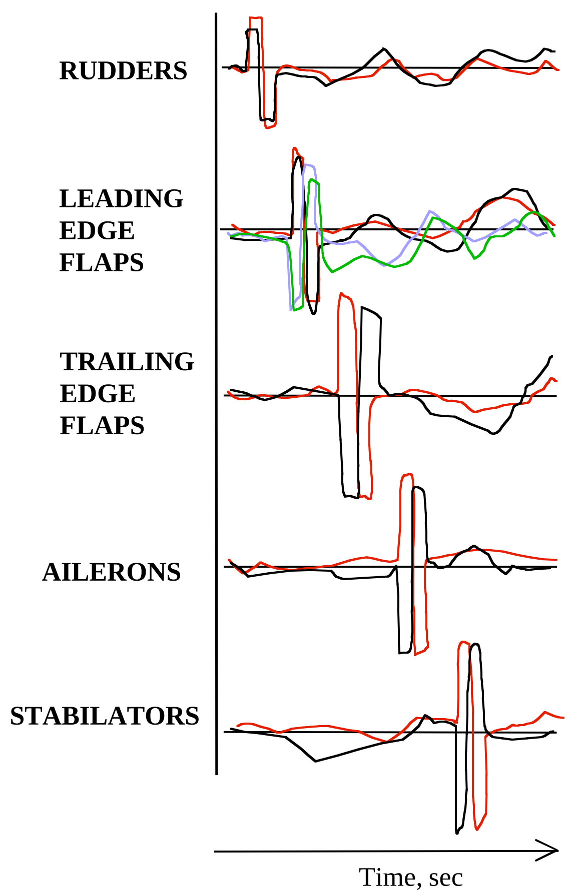
•Reduced risk associated with the Active Aeroelastic Wing program

•Increased model fidelity within the Active Aeroelastic Wing regions

Status

The flight test program is over, and the remainder of the data reduction is being performed. F/A-18 models for aerodynamics and loads are being refined based on the flight test data.

F/A-18 Test Vehicle



Flight Doublet Time History

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X-33 Avionics Integration & Real-time Nonlinear Simulation

Summary The X-33 program will prototype new technologies required for design of a Reusable Launch Vehicle (RLV). The X-33 will be an unmanned vehicle, launched vertically, reaching over 200,000 feet altitude at speeds approaching Mach 10. The vehicle will operate autonomously from launch to landing. Some of the technologies to be demonstrated are: metallic thermal protection system, linear aerospike engines, and an integrated vehicle health monitoring system. One of the ways that NASA Dryden is supporting this activity is through the development of an X-33 Integration Test Facility (ITF) lab.

Objective The ITF lab provides an essential role in the avionics systems development of the X-33. The initial phases have included the development and test of a real-time, nonlinear six-DOF simulation, that supports ground operations as well as all phases of flight. Activities are underway in the integration of flight avionics systems into a hardware-in-the-loop (HIL) simulation which will be used for Verification and Validation (V&V) testing of the avionics system.

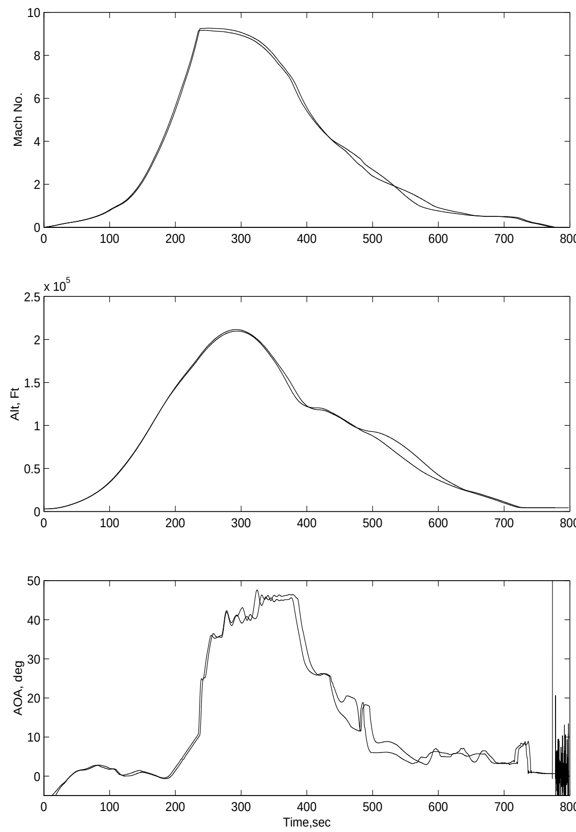
Approach The development approach begins with simulation models for the X-33 vehicle, e.g., aerodynamics, reaction control system, actuators, engine, navigation, guidance, and control. Each model is incorporated into the X-33 batch and real-time simulations. Six 1553 buses have been integrated into the real-time simulation to provide communication with the avionics subsystems. Once all of the hardware is fully integrated into the simulation and the final operational flight program (OFP) is delivered, formal V&V testing will be performed to certify the system is ready for flight.

Status Current work continues to be focused on model updates and hardware integration to support full, ascent - rollout, hardware-in-the-loop capability. Vehicle System Check Out (SCO) of the avionics subsystems are also getting much attention in the lab as these subsystems are installed on the vehicle.

Several elements of flight hardware have been and are being integrated at the ITF lab. These include Vehicle Mission Computers (VMC's), Flush Air Data System (FADS) Remote Pressure Sensor (RPS) units, INS/GPS units, and Forward, Rear, and Engine Data Interface Units (DIU's). The real-time VMC HIL simulation is now capable of flying X-33 missions from launch to landing.

Recent work has started in testing system robustness and stability margins of the flight control system. The robustness analysis is conducted through numerous Monte Carlo runs which are used to determine sensitivity to dispersions in critical simulation models, such as aerodynamics, engine, and vehicle mass properties. The stability margin analysis utilizes linear models of the vehicle dynamics combined with Matlab models of the control system. The linear models are derived directly from the six-DOF nonlinear simulation.

An X-33 Vehicle Mission Computer (VMC).



Comparison of HIL and SIL X-33 six-DOF simulated flight from liftoff at Edwards AFB to landing at Michael site in Utah.

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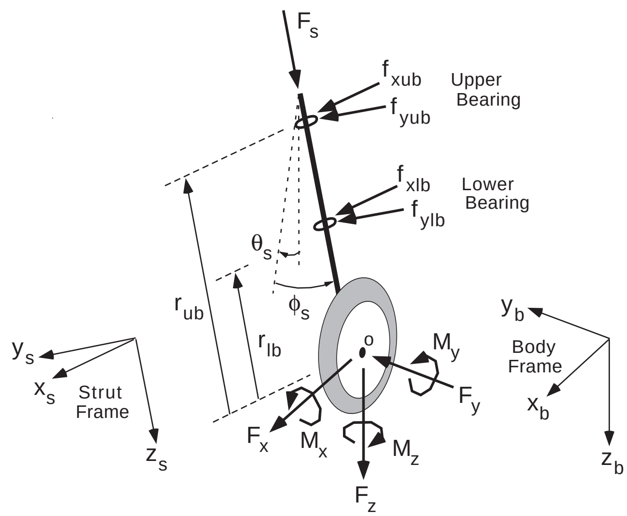
X-33 Landing Gear Dynamics Model

Summary: One of NASA Dryden's main roles on the X`33 program is to develop a high fidelity X-33 simulation to be used in verification and validation testing of the avionics subsystem. The most benefit from V&V testing is gained when the subsystem models provide the most accurate possible representation of the actual subsystems. The landing gear model must faithfully portray tire friction and compression as well as strut loads, compression and dynamics. This will indicate to V&V engineers the potential for gear failure, tail scrape or a dynamic interaction with the automatic flight control system as well as prove the performance of the ground steering control laws.

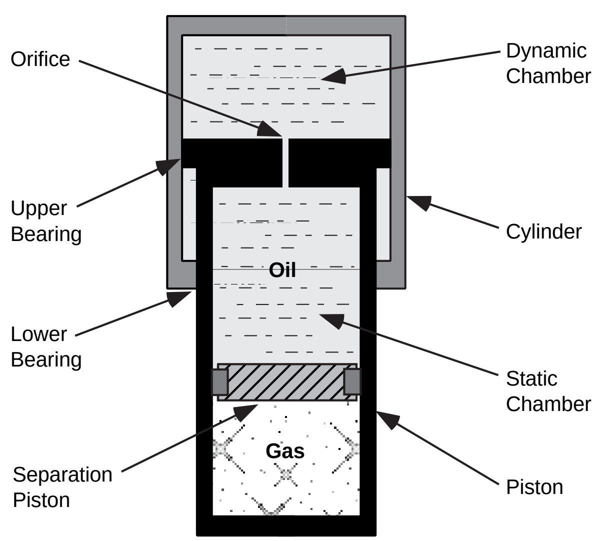
Objective: The objective is to develop an accurate landing gear model which may be easily integrated into the X-33 simulation and create the smallest possible computational burden. This will allow for real-time V&V simulation testing of the landing system produce high confidence in the results.

Approach: The core gear dynamics model evolved from a high fidelity landing gear simulation created by Lockheed Martin for the F-117. This core was modified with X-33 landing gear parameters and the equations of motion were expanded for greater accuracy.

Status: The landing gear model is installed in the X-33 six-DOF simulation. Also integrated into the gear dynamics model are models for the nosewheel steering controller and brake control unit. Current work is being performed to incorporate runway crown effects into the model and there will be adjustments made on some gear parameters to match drop test results.



Landing gear forces and moments.



Oleo-pneumatic strut.

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X-43A Non-Linear Monte Carlo Analysis

Summary

The Hyper-X program was conceived in order to demonstrate the first ever operation of an integrated airframe-scramjet vehicle in flight. The X-43A is the first in a series of such aircraft. It is carried to the test point, which is approximately Mach 7 at 95,000 ft, by a modified Pegasus launch vehicle. Following separation from the Pegasus, the scramjet engine is fired for a short period of time. Parameter identification maneuvers are then conducted until splashdown in the Pacific Ocean. Since this is the first in a series of unique missions, there are many uncertainties. Aerodynamic performance can differ from CFD and wind tunnel derived models by up to 30%. Actuator, mass properties and sensor uncertainties also have to be taken into consideration. The Hyper-X Controls group has developed a Monte Carlo simulation tool for the use of the Hyper-X project to demonstrate desired controller performance in the presence of modeling and measurement uncertainties.

Approach

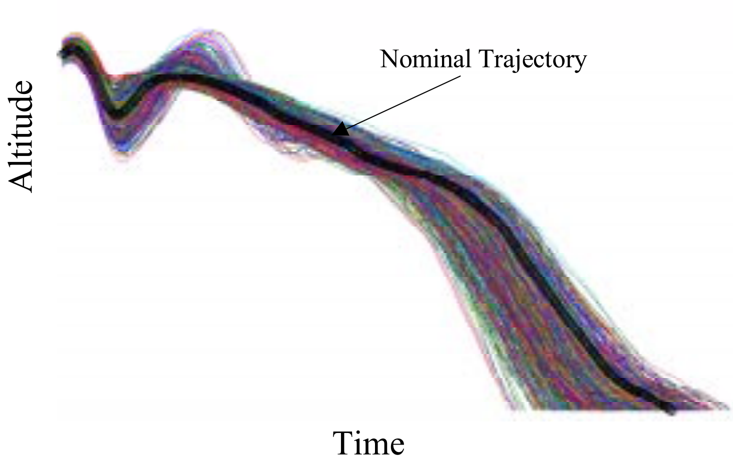
The Monte Carlo tool consists of a modified non-linear FORTRAN simulation of the X-43A and MATLAB scripts which are used to do the following: drive the non-linear sim with various uncertainties, analyze the results, save the data, and plot the data for further analysis. The uncertainties used consist of uniform or Gaussian distributions for atmospheric properties, actuator performance, aerodynamic characteristics, mass properties, sensor uncertainties, flight condition at separation, and separation forces and moments.

Results

The Monte Carlo tool has proven invaluable in confirming controller performance when the aforementioned uncertainties are applied. The controller in all cases achieves the requirement for a constant angle of attack during the scramjet test. The data has also been used to develop a trajectory footprint and understand the effects of various uncertainties upon the mission profile. Specific sim scripts generated for the Monte Carlo analysis have been selected for use during hardware in the loop validation testing in order to fully exercise the system within the expected mission profile.

Status/Plans

Model updates are added to the Monte Carlo tool, as they become available. 1,000 runs are generated every weekend to provide additional data for analysis. The tools continue to evolve as more uses are found for them.



Descent Trajectory

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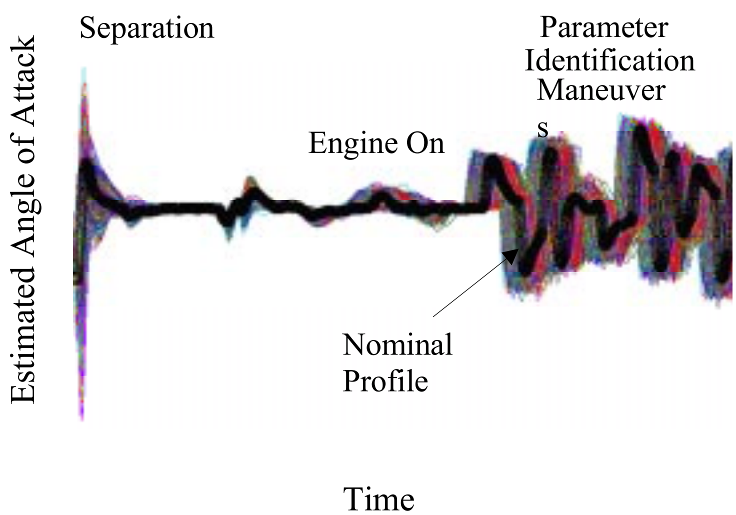

Alpha Profile Immediately after Separation

Unmanned Combat Air Vehicle

Summary
The unmanned combat air vehicle (UCAV) is a Boeing Phantom Works concept for a fully autonomous airplane designed to perform military suppression of enemy air defense (SEAD) missions. Work is currently being done to develop two UCAV demonstration vehicles, which will serve as a prototypes for the eventual operational vehicles. The demonstration vehicles will be required to taxi in formation, takeoff, fly in formation, execute a collision avoidance maneuver and land in a fully autonomous manner. Operations will be conducted under the supervision of a human operator who will monitor vehicle health, send basic commands and serve as an interface to air traffic control. Dryden will host the flight test program, which is expected to begin in December, 2000.

Objective
NASA Dryden's major roles in the program are to develop the automatic taxi control laws, and algorithms for formation taxi and collision avoidance. These products will be integrated into the vehicle control system and demonstrated during flight test.

Approach
Boeing has provided Dryden with a detailed, high fidelity Matrix-X 6-DOF simulator. All algorithms and control laws will be designed and validated using this single tool. Ultimately, the flight control software will be obtained by auto-coding the control law block diagrams directly from the simulator.

The auto-taxi control makes use of the throttle, collective and differential brakes, yaw thrust vectoring, speed brakes and nosewheel steering to steer the vehicle along a predetermined taxi path consisting of surveyed waypoints. The control laws are designed using nonlinear inversion techniques similar to the flight control laws. This will enable a smooth integration into the flight control system and minimize potential problems during transition to and from flight.

Dryden's UCAV team is also evaluating concepts for the integration of UAV aircraft such as the UCAV into airspace with piloted aircraft. Of primary concern is the ability of the UAV to "see and avoid" other aircraft. The Traffic Alert and Collision Avoidance System (TCAS), currently approved by the FAA for commercial transport aircraft, will be integrated with autonomous collision avoidance algorithms developed by MIT's Lincoln Laboratories onboard one of Dryden's high-performance research aircraft to demonstrate autonomous collision avoidance. These algorithms, using simulated TCAS hardware, will also be flown on the UCAV vehicle for a collision avoidance demonstration.

Artist conception of UCAV

Status The auto-taxi control law architecture is established and integrated into a local copy of Boeing's 6-DOF simulator. Formation taxi algorithms have also been developed. Current work is focused on refining the control system gains and integration with the flight control system.

The collision avoidance effort is currently procuring hardware and developing algorithms to be used to test and demonstrate a collision avoidance maneuver for UCAV.

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X-43A Hypersonic Blended Inertial and Pneumatic Angle of Attack

Summary

The Hyper-X research program, conducted jointly by NASA Dryden and NASA Langley, was conceived to demonstrate a scramjet engine in a flight environment. The X-43A Research Vehicle, the flight test instrument of the Hyper-X program, will be lofted to its pre-determined research test condition with the aid of a modified Pegasus first-stage booster. After separation from the launch vehicle and during the engine test phase, the X-43A will be commanded to follow a nearly ballistic flight path due to mass flow and angle-of-attack requirements imposed by the scramjet engine test. Key to mission success is the ability to accurately measure and control angle-of-attack.

Angle of Attack derived from inertial sources can provide adequate frequency content for flight control, but absolute accuracy is heavily dependent on an accurate atmosphere model – including winds. A Flush Airdata System (FADS) can provide highly accurate flow angles in the presence of winds, but significant pneumatic lag often limits the usefulness of this signal for real-time feedback. In order to provide an accurate, real-time angle-of-attack signal for Hyper-X flight control, a FADS derived angle of attack can be blended with an INU-derived angle of attack.

Objective

Dryden efforts over the past year have focused on improving the FADS algorithm performance with regard to angle of attack requirements for the scramjet engine test phase. Wind-tunnel data was used to calibrate the FADS angle of attack from Mach 3 to Mach 8. Significant effort was put toward developing and validating a simplified pneumatic lag model in order to compensate for the lag in the flight control laws. For simplicity, a first-order bias filter was selected to blend the FADS signal with the INU-derived angle of attack figure(1).

Results

A dynamic FADS simulation model was developed from static and dynamic wind tunnel test data. The simulation model has been used to generate Monte Carlo uncertainty results. Two project reviews of the blended angle-of-attack algorithm were conducted in FY99. Updates to the real-time angle-of attack estimation algorithm have been incorporated into the flight software release.

Figure 2 shows differential FADS pressure results from wind tunnel and simulation data for an angle-of-attack sweep from –6.5° to 12° at Mach 8. One key result from this research is that the pneumatic lag model for a pair of differential ports does not change significantly over the desired angle-of-attack range. This result is evident from the inserts shown in figure 2, showing similar differential pneumatic lag response to 1° steps in angle of attack throughout the test range.

Status/Plans

Hardware-in-the-loop and on- aircraft testing will be used to validate system implementation.
First flight of the Hyper-X is scheduled for summer 2000.

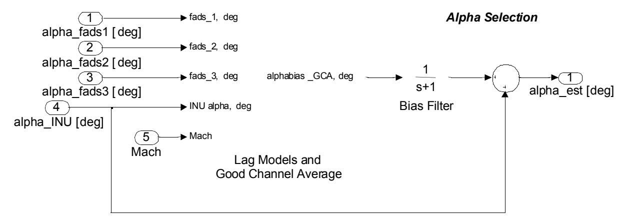

Figure 1: Block diagram for a blended FADS and inertial angle of attack.

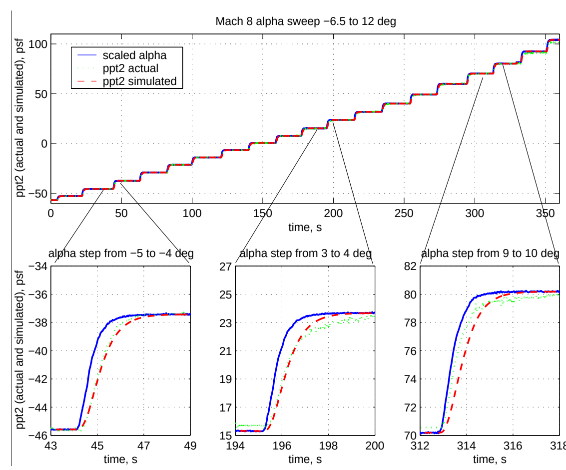

Figure 2: Mach 8 angle-of-attack sweep (AEDC wind tunnel data).

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F/A-18 Autonomous Formation Flight

Summary

Aircraft flying in formation can take advantage of each other's wingtip vortices to improve the overall formation efficiency. Two aircraft flying wingtip-to-wingtip can achieve the performance of a single aircraft with twice the aspect ratio. While wingtip-to-wingtip flight is unrealistic, this effect dissipates slowly with longitudinal separation. Two aircraft flying with wingtips aligned and within two or three wingspans longitudinally of each other can achieve significant drag reduction.

Objective

Demonstrate through flight test a formation flight autopilot system. This system will include all of the navigation, guidance and control functionality required to accomplish autonomous station keeping of one follower aircraft to a second leader aircraft.

Justification

Formation flight drag reduction has the potential to save commercial air cargo companies millions of dollars per year in fuel costs. This research also has benefits in the areas of Uninhabited Combat Air Vehicle (UCAV) formation flight, swarming and autonomous refueling.

Benefits

- Reduced drag and extended range
- Autonomous station keeping
- Cooperative formation control

Approach

A NASA F/A-18 chase aircraft will be outfitted with a formation flight autopilot system, comprised of a GPS receiver, an Airborne Research Test System (ARTS) and Production Support Flight Control Computers (PSFCCs). Navigation and guidance algorithms have been developed at Dryden along with outer-loop control laws to complete the formation autopilot system.

This specially modified aircraft will fly in formation under autopilot control with NASA's F/A-18 Systems Research Aircraft (SRA) in flight tests scheduled for the summer of 2000. The aircraft will maintain a separation of 250 feet to ensure safe operations.

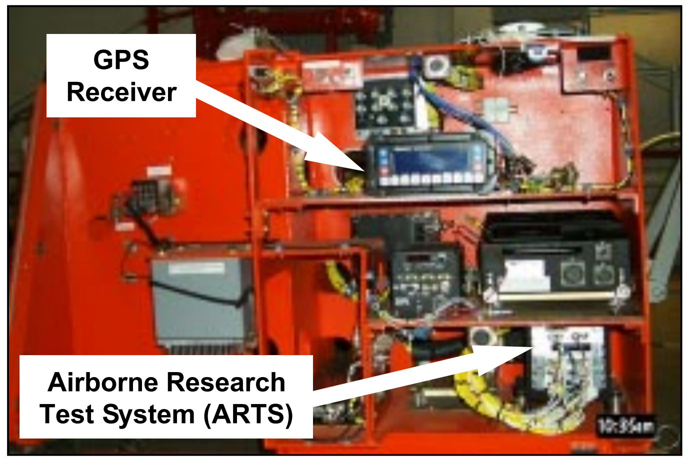
Results

The GPS relative navigation system has undergone risk reduction testing using NASA's King Air aircraft. These tests have demonstrated that two independent GPS systems on the same aircraft provide a relative position accuracy of approximately 1.5 feet. A relative position accuracy of 15 feet or better is required for successful formation station keeping.

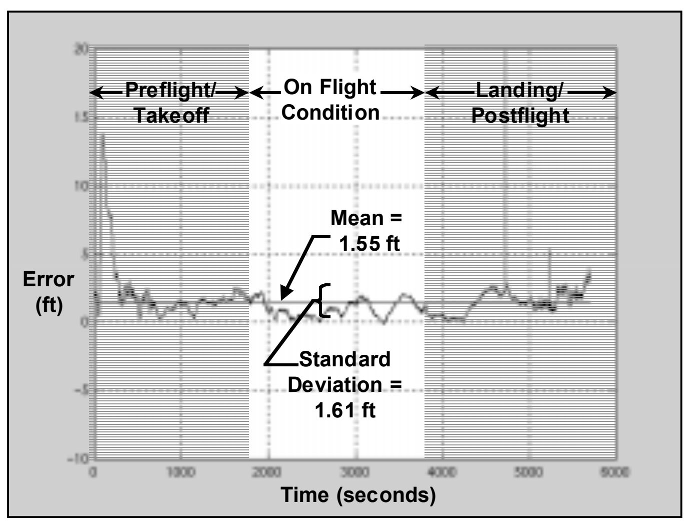
Status

Final software and hardware modifications are being completed in preparation for verification and validation testing of the formation flight autopilot system. Risk reduction flights to further evaluate the GPS relative navigation system are planned for early 2000. A software in the loop simulation that will facilitate control law evaluation and flight test planning is nearing completion.

Flight test of the Autonomous Formation Flight system is scheduled for June of 2000.



Instrumentation Pallet Containing Components of the Formation Flight Autopilot System



Relative Position Accuracy of Two GPS Antennas in Flight at a Fixed Separation

Contact

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F/A-18B Systems Research Aircraft (SRA)
X-33 Vehicle Health Monitoring System Risk Reduction Experiment

Summary

A VME Based VHM computer, remote health nodes (RHN), and Distributed Strain Sensor (DSS) module were installed and integrated onto the SRA in September of 1999. A Structures Flight Test fixture was added to the experiment in November of 1999 to support the DSS flight research efforts. The fixture was removed in Early December of 1999 and the aircraft was returned to the VHM computer and health node configuration. The experiment has flown for 34 flights, six of which were flown as dedicated DSS research flights with the flight test fixture installed on the aircraft. Data reduction and analysis is ongoing.

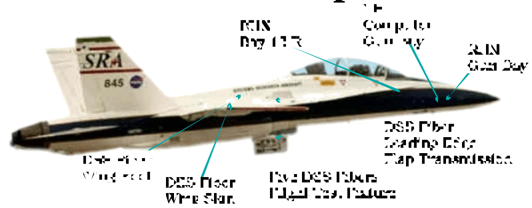
Objectives

This experiment was designed to demonstrate that the VHM system could be successfully integrated and flown on a high performance aircraft and to validate fiber optic strain measurements against conventional technology when acquired in a realistic flight environment. The specific objectives of the experiment included.

- Demonstration of the installation and performance of the Generation II Multi Mode fiber optic cable plant.
- Demonstration of the capabilities of the RHN and validate the accuracy of the measurements.
- Verification of the installation and integration procedures of the VHM system on the aircraft.
- Validation of the correlation between the fiber optic DSS and data recorded using conventional strain sensors.
- Provide a configuration for ongoing VHM research efforts.

Justification

The realization of integrated health management (IVHM) involves acquisition, distribution, analysis, and decimation of vehicle information to the user. This involves fusing new or existing sensor technology with acquisition and distribution devices then implementing prognostic algorithms, and displaying the resulting information to the correct person in a timely manner. This experiment applies both new and existing sensor technologies with a new data distribution network. Components of the system have been tested in a laboratory environment, but have not been demonstrated in a flight environment.



Approach

The VHM computer and one RHN were installed in the Gun Bay on the SRA. A second RHN was installed in Avionics Bay 13 Right. These nodes collected analog sensor data and transmitted the information to the VHM computer using a multi mode Generation II fiber optic cable plant. The DSS experiment consisted of a tuneable laser and signal processing module, located in the VHM computer. Eight single mode fibers and Bragg Grading were installed on the aircraft and flight test fixture (three on aircraft structure and five on the flight test fixture.) to measure strain in a flight environment. Data from over thirty-four flights have been collected using the VHM computer and RHN's. Six flights were dedicated to flying the DSS experiment and flight test fixture. Flight test fixture flights were flown to 100% of the design limit load (DLL) on the fixture.

Results

- Installation and integration of a generation I VHM architecture consisting of a Distributed COTS Based, Open Architecture design concept.
- Flight demonstration of a Generation II Multi Mode Fiber Optic Cable plant transmitting data between health nodes and the VHM computer.
- Flight demonstration of a network of fiber optic distributed strain sensors.
- Integration and flight test of a generic flight-test fixture to 100% of DLL.

A NASA paper reporting on the results of the experiment is being initiated.

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Flight Test of an Intelligent Flight Control System on the F-15 ACTIVE

Summary

The F-15 Advanced Controls Technology for Integrated Vehicles (ACTIVE) was the testbed for flight test of the Intelligent Flight Control System (IFCS). The IFCS is a flight control concept which utilizes a Neural Network (NN) to determine critical stability and control derivatives for a control law that calculates real time feedback gains with a Ricatti Solver. These derivatives are also used for plant identification in the model following portion of the controller to insure level one handling qualities. The Flight Test of the IFCS was the culmination of the joint Intelligent Flight Control Advanced Concept Program teaming NASA with Boeing Phantom Works. The goals of the IFCS Advanced Concept Program (ACP) are:

1. Develop a Flight Control Concept that uses Neural Network Technology to identify aircraft characteristics to provide optimal aircraft performance.
2. Develop a Self-Learning Neural Network to update aircraft properties in flight.
3. Flight demonstrates these concepts on the F-15 ACTIVE Aircraft.

F-15 ACTIVE

Objective

The IFCS Advanced Concept Program objectives consist of three phases:

Phase I: Pre-Trained Neural Network Development.

Phase II: On-Line Neural Network Development.

Phase III: Neural Network Flight Controller Development.

Approach

Flight test of the IFCS ACP was performed in 3 stages. Phase I & II was flown with system outputs provided to instrumentation only and was not used for aircraft control. Phase III utilized the Phase I Pre-Trained NN to provide real-time aircraft stability and control derivatives to a Stochastic Optimal Feedforward and Feedback Technique (SOFFT) controller developed by NASA Langley. This combined Phase I/III system was flown utilizing the F-15 ACTIVE Research Flight Control System (RFCS). The RFCS allows the pilot to quickly switch from the experimental research flight mode back to the safe conventional mode if required.

Results

IFCS ACP flight test was completed in April, '99. The Phase I/III flight test milestone was to demonstrate across a range of subsonic and supersonic flight conditions that the Pre-Trained Neural Network could be used to supply real-time aerodynamic stability and control derivatives to the closed-loop optimal (SOFFT) flight controller. Additional objectives achieved include:

1. Flight Qualifying a NN Control System.
2. Use a combined Neural Network/closed-loop Optimal Flight Controller to achieve Level 1 flying qualities.
3. Demonstrate, via Dial-a-Gain, that different flying qualities could be achieved by setting new target parameters ω_{sp}, ζ_{sp}, ω_{dr}, ζ_{dr} and t_r.

In addition, data for the Phase II (On-Line Learning NN) was collected using AdAPT (Stacked Frequency Sweep) Excitation for post flight analysis. Initial analysis of this data showed good potential for real-time identification and online learning.

Reference

Intelligent Flight Control: Advanced Concept Program, Final Report, Boeing-STL 99P0040 (NAS2-14181)

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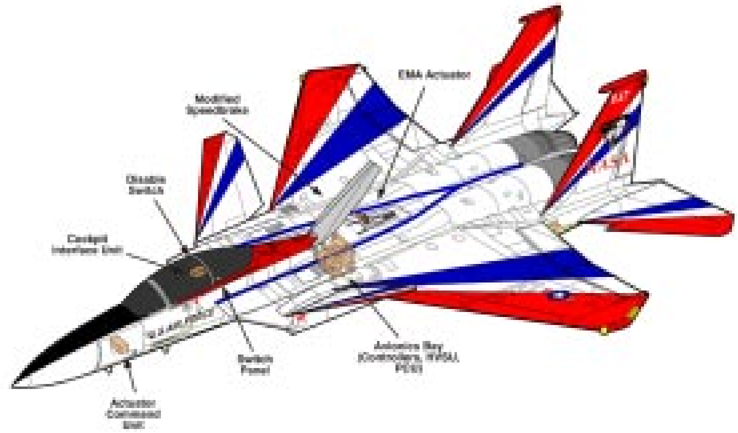
X-Actuator Control Test (X-ACT) Program

Summary

For X-ACT, the X-38 EMA will be installed in place of the standard hydraulic speedbrake actuator on NASA #837, a heavily modified, preproduction F-15B two seat fighter aircraft. An Actuator Control Unit (ACU) will be used to provide actuator commands normally generated by the X-38 Flight Critical Computers. Actuator commands will be specified by a Flight Test Engineer in the back seat of the aircraft via a cockpit interface system. The X-38 Failure Detection, Isolation, and Recovery (FDIR) logic will be hosted in the ACU, with full authority to reconfigure the system. Failures can be injected via the crew interface, either by modifying data passing between the ACU and actuator controllers, or by interrupting signals between the actuator and controller.



X-ACT System Locations

Objective

The primary technical objectives of the X-ACT program, can be grouped as follows:

Actuator Integration and Flight (JSC)

Uncover unanticipated problems in taking the system to flight, validate/develop an actuator model under actual flight conditions, design and integrate a high power, multichannel EMA system onto a testbed aircraft, and evaluate EMAs under specific simultaneous load, rate, and extension conditions in a flight environment.

Redundancy Management System (JSC)

Design, code, implement, and test the redundancy management software for the X-38/CRV EMA system and validate redundancy management software using a fault injection system.

Cockpit Interface System (DFRC)

Development of a Multipurpose Cockpit Display System for Flight Research., develop graphic display software for onboard monitoring of research experiments, including mission critical parameters, develop graphic interface software to control onboard research experiments, develop a flexible command profile generator system, and design, implement and test a hardware and software fault injection system.

Justification

The research value of the X-ACT flight test program is in the establishment of a new redundancy configuration, potentially suitable for a wide range of new vehicle applications. These flight tests will either verify the correctness of the design approach or identify deficiencies in the configuration to allow correction. Lessons learned during this program should be applicable to other high power EMA and EHA programs currently in development, both for spacecraft and aircraft applications. A successful flight test of the actuator could help further the acceptance of Power-By-Wire actuation technology by industry.

Approach

The flight test matrix will span the rod end load and rate envelope of the actuator. The actuator will be tested throughout the full range of its stroke. Some typical X-38 reentry profiles may be flown, from an altitude of 50,000 ft. down to the moment of X-38 parachute deployment. The goal is to have 25 flight hours on the actuator by the time vehicle V-201 ships to KSC for Shuttle Integration. An additional 75 flight hours will be accrued prior to first flight of the CRV vehicle.

Results

- Currently under development

Benefits

- Reduced Risk for X-38.
- Validation of new Redundancy Management Schemes for Power-By-Wire Actuators.
- Lessons Learned applicable to multiple EMA development programs.

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X-43A ANTENNA PATTERNS

Summary

The Hyper-X research program, conducted jointly by NASA Dryden and NASA Langley, was conceived to demonstrate a scramjet engine in a flight environment. The X-43A Research Vehicle, the instrument of the Hyper-X program, will be lofted to its pre-determined research test condition with the aid of a modified air-launched Pegasus booster. After separation from the launch vehicle and during the engine test phase, the X-43A will be commanded to follow a nearly ballistic flight path - a result of scramjet engine angle-of-attack requirements. The engine test phase (which includes post-test vehicle parameter identification maneuvers) is concluded by a recovery to a nominal descent trajectory made possible by the autonomous controller resident in the vehicle's flight control computer. Since the vehicle is autonomous the only link to ground is by transponder and telemetery. The telemetry system is transmit only. The transponder operates as a beacon. When queried the transponder will transmit. Several factors depend upon the reception of data transmission from an aircraft. One is the installed antenna pattern of the aircraft. The X-43A has three antennas, two on each side and one facing the rear. Knowing the transmit pattern of the antennas will provide the ability to receive the strongest transmission of radio frequencies from the aircraft.

Objective

The overall test objective was to gather qualitative antenna pattern data from installed antennas on the HXRV. This objective was accomplished by collecting antenna patterns at selected elevations of the HXRV. The test data will be used to verify adequate (radio frequency) RF coverage at possible ranges and altitudes the aircraft will be located during its flight. The data will identify null areas of the RF coverage and assist in orienting the flight path and airborne and ground assets for best RF reception.

Justification

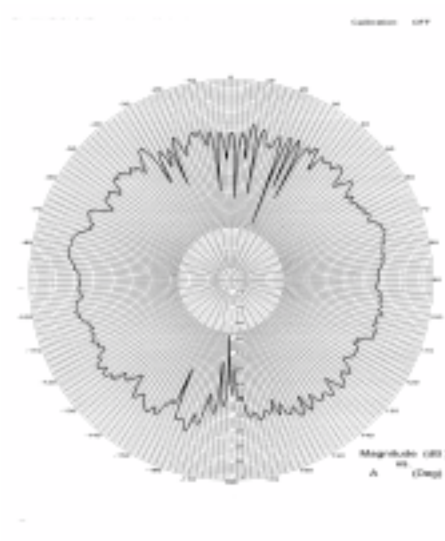
Antennas installed on aircraft usually have a much different pattern compared to uninstalled patterns. The airframe may act as an electrical conductor that may transmit electromagnetic energy throughout the structure. Other aircraft structures may block the magnetic energy. The antennas depending upon location may interfere with each other. These and other electromagnetic properties enter into the radiated pattern of the antennas.

Approach

The X-43A aircraft was placed in the Benefield Anechoic facility at Edwards AFB, California for testing. The aircraft exterior was kept as close as possible to the flight configuration. The internal structure was the hull without the various flight hardware. The aircraft was lifted by hoist to an elevation that may be seen in flight. A network analyzer generated various frequencies to radiate at the aircraft. The aircraft was tied to a positioner to rotate the aircraft 360 degrees. These frequencies were received by the aircraft antennas and sent to an RF receiver. The receiver was connected to a plotter that provides antenna polar plots.

Results

The antenna plot below provides a tangible picture of what the three antennas would receive or transmit during flight. The plot is representative of the electromagnetic environment around the aircraft and the aircraft's transmit and receive capability at the specific RF frequency and at the elevation of the pattern.



X-43A Antenna Plot

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Reconfigurable VHF Broadcast Differential GPS Base Station

Summary

Differential Global Positioning System (DGPS) has been identified as a requirement for an increasing number of flight projects. This technology allows positioning of aircraft in real time to within a few meters or centimeters. Rather than each project putting together their own ground station, a common resource would provide a standard and satisfy many project requirements, yet be reconfigurable for custom requirements.

Objective

Design, develop, and demonstrate a real-time, reconfigurable broadcast DGPS system to achieve sub-meter aircraft positioning in real time.

Justification

While various types of DGPS ground stations exist, they are normally configured in a static configuration and often not optimized for precision flight research work. Flight research often requires a reconfiguration of support systems, including DGPS. Reconfiguration requirements may include changes to frequencies used, data rates, data formats, message types, transmit power, and date logging. Having a NASA controllable asset that can be scheduled specifically for particular configurations will help NASA Dryden to support a variety of GPS activities.

Approach

A dual set of Ashtech Z-12 GPS receivers were purchased, along with sets of error-correcting radio modems and antennas. This equipment was installed in a building near the main runway so as to optimize both visibility while in the pattern and visibility within a 200 mile area surrounding the airfield. Either standard RTCM SC-104 code phase or DBEN carrier phase (Ashtech proprietary) messages can be transmitted to aircraft containing radio modems and GPS receivers--allowing aircraft position reporting in the meter (using RTCM) or centimeter (using DBEN) ranges.

Results

The system is in place and transmitting. It is being used initially to assist in understanding X-33 related GPS systems and is ready to support autoland and autonomous formation flight (AFF) activities, as well as X-34 activities, if required.

Benefits

- Enhances Dryden's range capabilities.
- Improves the accuracy of on-board GPS data.
- Provides a basic GPS functionality that can be modified when requirements change.
- Increases personnel understanding of DGPS.

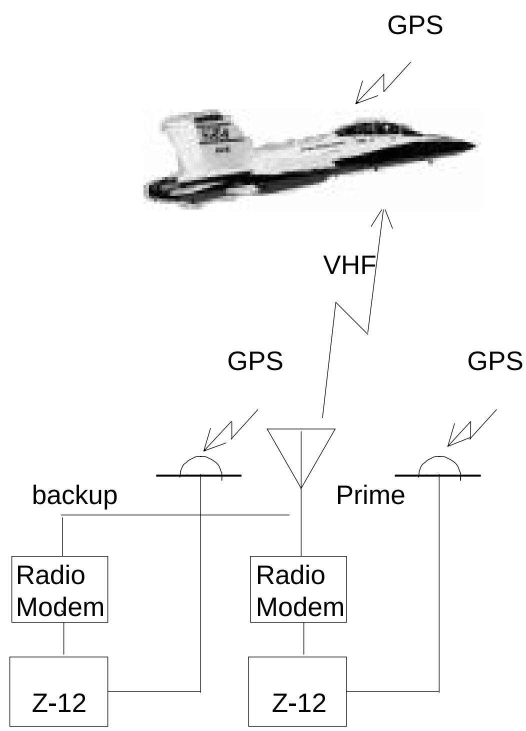

Diagram of DGPS ground station and application.

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DSP/GEDAE Signal Processing Algorithms

Summary

The understandings of dynamic phenomena are often essential in the development of modern aircraft systems. To describe these, one may use either a statistical characterization of the signal, such as standard deviation or higher moments (skewness or flatness) or zerocrossings. These are distinct parameters that fit well in the PCM stream.

However, if spectral information is required it is necessary to compress the data either accepting some loss in information, or if processing speed and memory buffer permits, do a lossless compression.

The system under development encompasses selective digital filtering rather than analog to increase flexibility, signal characterization through statistical parameters to allow parameter identification and spectral analysis/compression to allow merging with lower frequency parameters.

The illustration shows a typical hot-film spectrum taken in flight that would require 1800 bits uncompressed. The information is transferred lossless using approximately 600 bits utilizing among other techniques bit-packing, spike removal along with inversion. If a small loss is allowable, floating point curve representations can be used, resulting in further reduction in bit transfer requirements.

Objectives

DFRC needs to improve ground and airborne capabilities to encode, record, process, correlate, and archive high frequency hot film and microphone research data. Traditionally most flight parameters are made available to the researcher in the form of time-series in the FDAS system, while the dynamic data has much higher bandwidth requirements. The objective of the present work is the development of a data acquisition subsystem which can process high frequency data in real time and output results that can be merged with slower PCM streams for display and archival.

Results

Ground System Development:

> Desktop analysis tools were evaluated and GEDAE was chosen. An embedded PC board running NT was chosen as the host system which can then configure the DSP to run the desired algorithm.

Software tool development:

> Smoothing filters, Integrating filters, Differentiating filters
> Spike removal, Curve fitting, deviations
> cross correlation

Ground system testing:

> Tools were tested and used against Hot Film data sets supplied by LaRC and data stored on FDAS

Status/Plans

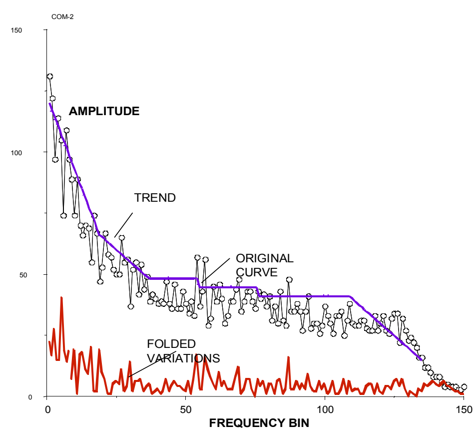
Define a flight system and the associated control room part of the system. Acquire flight hardware.

Perform statistical and spectral analysis to reduce the data so that it can be merged with traditional PCM streams.

Continue to integrate tools onto the embedded DSP board.

Work directly with Hot Film sensors and the IXTHOS Sharc DSP.

Spectrums from hot film sensor in flight at hypersonic



Mach number. Example of loss less compression.

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Multiplexed Hot Films

Summary

Surface-mounted hot-films are commonly used to detect transition from laminar to turbulent flow and separated flow regions on wind tunnel models as well as in flight. With the introduction of high-speed data acquisition and processing it becomes feasible to document the flow using sheets of sensors arranged in arrays, which are switched in as needed.

An effort to explore the flow physics of a three-dimensional high-lift configuration in a wind tunnel is planned at NASA Langley. Approximately 650 films will be used in various combinations depending on real-time decisions when various flow phenomena are encountered. These can be attained through software-controlled, commercially available switches. A total of 24 temperature-compensated constant temperature bridges are used, organized in three modules of 8 channels each. NASA Dryden-developed anemometry is used to run the films, taking advantage of the size, simplicity, robustness and low noise level. Both power to the bridges as well as signals are switched.

To evaluate the design of the transition experiment in terms of ability to identify transition/separation features and support the software development as well as learning about the flow physics, two risk-reduction experiments were conducted in Langley's Basic Aerodynamics Research Tunnel (BART). The system turned out to be very reliable, both in terms of the sheets with sensor arrays, and the functionality of the anemometer circuits and the switches.

The utility of the modular design was demonstrated when hot films were needed for exploration of various models of the 2003 Mars Airplane. Depending on the required number of films, anticipated flow conditions, scheduling etc., the system was split into several subsystems, each with anemometer modules and switches. The data acquisition codes required minor changes and the self-descriptive file system definition originally developed for the high-lift experiment was used.

Objective

Dryden efforts are focused on making a system available "on the shelf", allowing quick response to needs that may arise. This involves mainly hardware and data-acquisition software as well as use of film arrays. Utilization of the current anemometer design together with flight-hardened switching units are short-term objectives. A more long-term goal is a revision of the anemometry itself to include analog-digital conversion, switching and optimization at the board-level.

Real-time processing, compression and merging of the high-frequency data are described in a separate highlight.

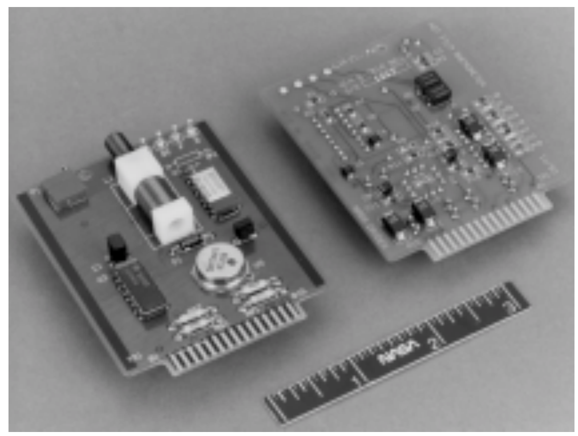
Results

So far the approach has been used in five successful wind tunnel experiments at NASA Langley. Dryden have participated through making the anemometry units available and working with Langley to develop a better understanding of system performance. Experimental results have been shared to better guide the post-processing and merging of the data with low-frequency data.

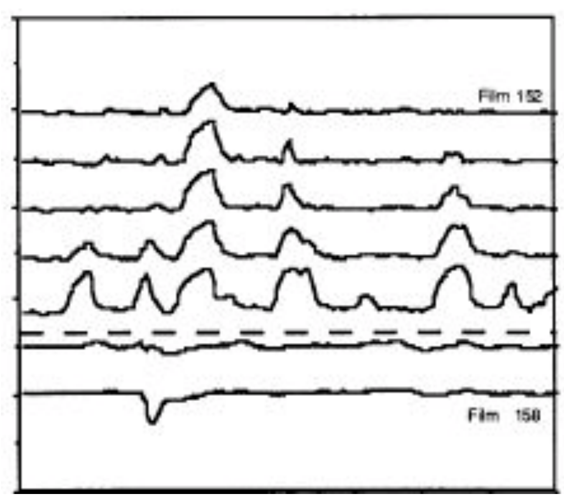
Status/Plans

The main outstanding issues as far as Dryden is concerned, is the flight-hardening of the switching units as well as the control of these for flight applications.

PHOTO OF UNITS



SAMPLE SIGNAL



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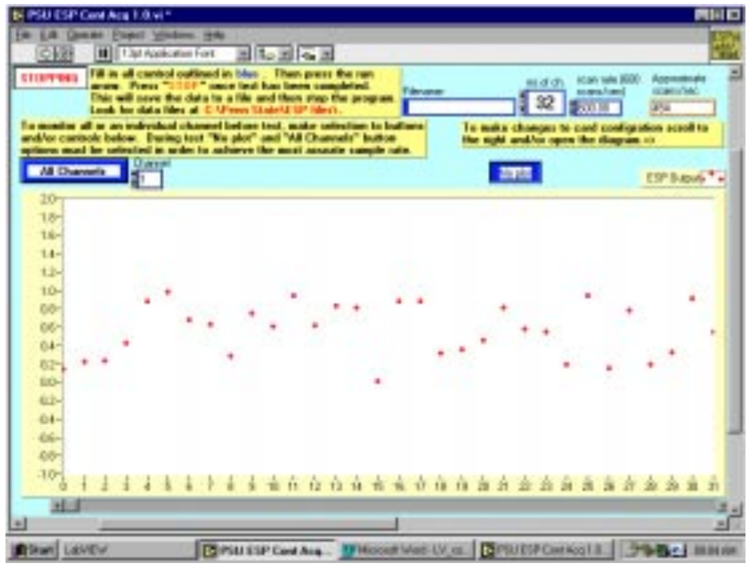
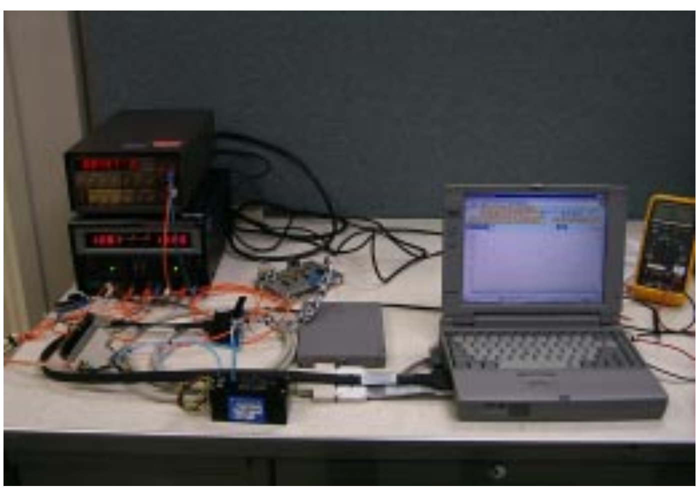
LABVIEW Based ESP Pressure Data Acquisition System

Summary

A supersonic wind-tunnel test was conducted at Penn State University to study shock/boundary-layer interactions. Multiple pressure measurements along a flat-plate test article were required to adequately characterize the pressure profile in the vicinity of an impinging shock. It was required to record 32 pressure measurements at 20 samples per second to capture the unsteady nature of the shock/boundary-layer interaction in the wind tunnel facility. Since a pressure measurement system that could meet these requirements did not currently exist at the facility, it was determined that NASA Dryden develop a low-cost and fairly "non-intrusive" pressure data acquisition system. This data acquisition system was to be completely separate from the wind tunnel data acquisition system with time tags correlating the two systems. Utilizing a laptop computer containing National Instruments hardware, LabVIEW software and an Electronic Scanned Pressure (ESP) 32 channel Pressure Transducer, a Pressure Data Acquisition System was developed. The test setup is shown in the photograph below.



LabVIEW program panel



Lab Test Setup

Objective

Develop a system that would allow the investigators to acquire pressure distributions without conflicting with the existing wind tunnel data acquisition setup and to provide a "user-friendly" interface.

Results

The LabVIEW based ESP Data Acquisition System created for the wind tunnel tests was developed at Dryden, shipped to Penn State, and successfully used to collect the desired pressure data. The system obtained data at a rate of 640 scan/sec reliably. This resulted in 32 pressure channel measurements at a rate of 20 samples per second. This data was then saved to a spreadsheet file.

The program had a pre-test functionality to verify that all 32 ESP channels were in working order. Selecting the chart option on the LabVIEW front panel, shown above gave a time history of the pressure measurements and could be viewed real-time on the screen. This reduced the sample rate and was therefore only used for pre-testing the sensor.

Status

This system has been developed and may be utilized in other tests where multiple pressure measurements are required and with minimal impact to the setup. The data acquisition system may also be used to test and verify ESP sensors on an aircraft or to ensure all ESP channels are in working order.

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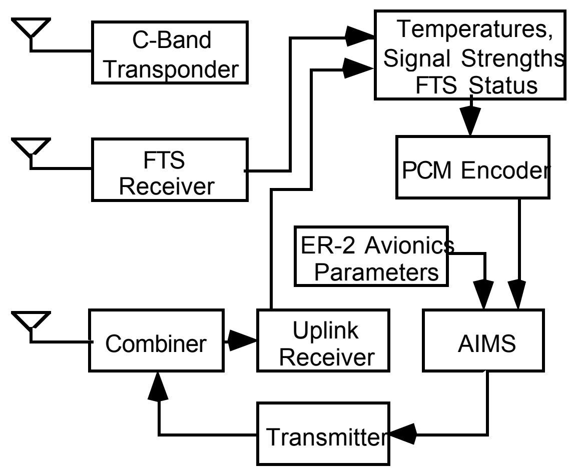
Flight Instrumentation for X-33 Extended Range Test

Summary
The X-33 project presents unique tracking requirements because the vehicle trajectory is beyond the line-of-sight of any single range. The X-33 is planned to fly between Edwards Air Force Base, California and Dugway Proving Grounds, Utah. An extended range was established to provide the continuous tracking necessary for maintaining public safety within the area effected by the X-33 trajectory. High altitude flight tests were required to confirm that the design, construction, function and performance of each extended range system satisfied the requirements.

Approach
A subset of X-33 avionics was integrated on an ER-2 aircraft in addition to a small instrumentation package. The X-33 avionics consisted of a C-band radar transponder and custom high-temperature flush-mount antenna, a Flight Termination System (FTS) receiver and stock antenna, a telemetry transmitter and custom, high-temperature flush-mount antenna, a hybrid RF combiner, and command uplink receiver. The Code RI-implemented instrumentation package consisted of DFRC signal conditioning, a PCM encoder, and a DFRC Airborne Instrumentation Management System (AIMS). The instrumentation package monitored avionics temperatures, signal strengths and FTS commands. The PCM multiplexer encoded these measurements in an IRIG format telemetry stream. The AIMS added ER-2 avionics parameters from a serial RS-232 format to the telemetry stream and output the data stream to the telemetry transmitter. Power to each subsystem was controlled by a series of fuses, circuit breakers, and relays. The relays allowed the pilot to simulate failures of various on-board X-33 range communication avionics subsystems.

Results
An instrumentation package and a subset of X-33 avionics used for range communications were successfully integrated on a NASA Dryden ER-2 for high altitude flight test. Uplink receiver signal strength levels met requirements. The requirement for continuous telemetry coverage from takeoff to landing was met, as was the requirement of at least a 6 dB link margin. Continuous FTS coverage and command integrity was verified. Integrated range systems operations and interfaces were validated, and operations training and practice for all range systems that will be utilized for X-33 operations were provided, including investigation of and training for failure modes and effects.

Status/Plans
Two more ER-2 flights are currently planned to re-validate X-33 Extended Test Range assets. These flights will take place approximately two months prior to the first X-33 launch. Before the next flights, it is planned to upgrade the instrumentation package with a higher bit rate PCM encoder. Also, the capability to wrap-around uplink commands is planned for the AIMS. Improvement of the wiring harnesses on the avionics pallet associated with both packages is planned.



Block Diagram

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Efficient Modulation Techniques

Summary

NASA Dryden and UCLA are investigating and testing efficient modulation techniques for use during flight- testing. The efforts include flight testing of Feher Quadrature Phase Shift Keying (F-QPSK)* modulation, research into other efficient modulation techniques, investigation of methods to improve the performance of F-QPSK as well as other modulation systems and research into the use of cellular phone Space-Time Coding methods for efficient data transmission.

F-QPSK modulation has been adopted by the Range Commanders Council for high data-rate telemetry systems and is capable of transmitting twice the data of a conventional Pulse Code Modulated/Frequency Modulation (PCM/FM) system in the same bandwidth, figure 1. Dryden is investigating the performance of sample F-QPSK systems through flight testing aboard the NASA King Aire aircraft. Dryden is also working with UCLA to investigate modulation techniques that may yield efficiencies greater than that of F-QPSK.

UCLA is currently investigating equalization methods to improve the current performance of QPSK systems, which could also be applicable to F-QPSK. The equalizer provides amplitude and phase compensation to restore the received signal to it's original transmitted form and reduce the effects of fading on the quality of received data.

UCLA has proposed the investigation of Space-Time Coding to increase the efficiency of QPSK, Quadrature Amplitude Modulation (QAM) and other modulation techniques. The Space-Time Coding technique utilizes multiple transmit and receive systems, figure 2, to improve bandwidth efficiency and Bell Laboratories has reported laboratory test results for cellular phone applications.

Objective

Current and proposed instrumentation systems require greater bandwidth for the transmission of test data than those used is the past. The bandwidth available for the transmission of flight test data is limited and methods need to be investigated to alleviate restrictions that limit the quantity and quality of data that can be obtained during flight testing. This research will allow for the transmission of instrumentation data at greater rates resulting in improved accuracy and performance.

Results

UCLA has completed an initial search of existing modulation techniques which have the possibility of yielding greater bandwidth efficiency.

UCLA has also completed the initial modeling and simulation of equalizers for QPSK systems as well as modeling and simulation of a Space-Time coding system using QAM and a Rayleigh fading channel model.

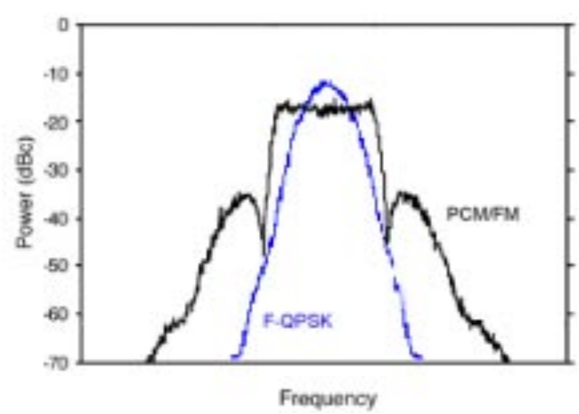
Status/Plans

Flight-testing and evaluation of F-QPSK system performance is planned for FY00. A PCM/FM system will be flown at the same time for comparison. Bit error rates and multipath performance will be evaluated and compared. Compatibility with existing ground station equipment will also be evaluated during integration.

Modeling and simulation of equalizers for various efficient modulation systems will be pursued with future efforts focused on equalizer optimization for Dryden's transmission channel.

Space-Time Coding system performance, for Dryden's fading channel, will be evaluated in future simulations to determine the applicability of this technique to Dryden's fading channel model. The use of Space-Time Coding for alternative efficient modulation techniques will also be the focus of future investigations. If Space-Time coding proves to be effective for Dryden's channel model it could be applied to improve the performance and efficiency of F-QPSK and other efficient modulation techniques.

Figure 1: Bandwidth of FQPSK vs PCM/FM



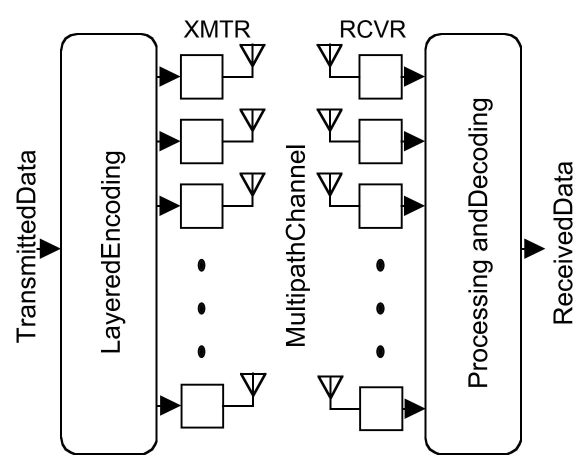


Figure 2: Space-Time Coding System

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* F-QPSK is Feher patented QPSK.

X-43A Flight Research Plans Overview

Summary The X-43A Flight Project includes the development and flight test of three hypersonic aircraft. Each fully autonomous vehicle is nominally 12 feet long, has a 5 foot wingspan, and is powered by a single airframe-integrated scramjet propulsion system. The first two vehicles will be flown at Mach 7 and the third at Mach 10. The research vehicle (HXRV) will be boosted to the desired flight test conditions by a modified Pegasus booster (HXLV). The HXRV will then separate from the HXLV, quickly stabilize, open the engine cowl, and perform a series of pre-programmed maneuvers to evaluate vehicle performance.

Objectives The overall program objectives include: (1) evaluation of the in-flight performance of an airframe integrated hydrogen fueled, dual-mode scramjet powered research vehicle at Mach 7 and 10; (2) demonstration of controlled powered and unpowered hypersonic aircraft flight; and (3) obtaining ground and flight data to validate computational methods, prediction analyses, test techniques, and operability that comprise a set of design methodologies applicable to future hypersonic cruise and space access vehicles.

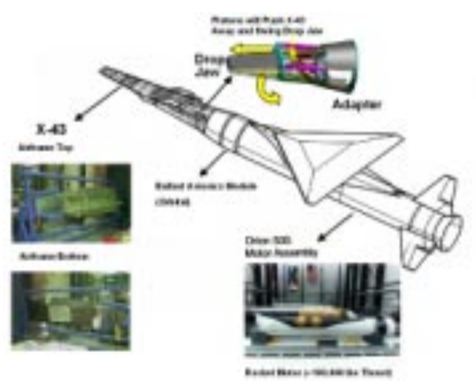
Specific flight research flight research objectives include development of in-flight performance analysis techniques, characterization and comparison of the predicted and flight aero-thermodynamic and propulsion data bases, validating the flight control law design and propulsion control law design, evaluation of a flush air data system, characterizing the performance of the vehicle systems, and characterizing stresses on key vehicle structures.

Justification The potential payoffs in a scramjet powered vehicle for access to space and/or high speed cruise has been well documented. The flight demonstration of an aircraft powered by an airframe integrated scramjet has never been accomplished and will add greatly to the confidence and credibility of airbreathing propulsion powered vehicle concepts.

Approach The approach from the beginning of the project has been to balance technical objectives, safety, and mission risk. To this end careful attention has been paid to hardware and software qualification and testing. A very large number of ground tests and analyses have also been conducted to better understand everything from engine performance and operability to the effects of side loads on the pistons used to separate the research vehicle from the launch vehicle.

Monte-Carlo analyses have played a critical role in understanding mission risk and maximizing mission success. At the backbone of these analyses are three highly sophisticated simulators corresponding to the three mission phases, namely launch, separation, and research vehicle free flight. Significant project resources have gone into the development of these simulators and the models and databases used by them.

Status Several significant milestones were achieved this year. Of particular note is the delivery of the first research vehicle and launch vehicle to Dryden, verification testing of all systems aboard the research vehicle, and completion of validation testing of the launch vehicle.



The Hyper-X Stack

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X-43 Fluid Systems Development

Summary

The X-43 autonomous, free flying, expendable research vehicles will conduct short duration scramjet flight tests at Mach 7 and 10. Each vehicle will be delivered to the test condition by modified Pegasus booster launched from a B-52 aircraft. The X-43 propulsion system consists of the scramjet engine itself, as well as the associated fluid systems. The fluid systems consist of the following. A high pressure gaseous hydrogen (GH2) blowdown system fuels the engine. A pyrophoric 20% silane - 80% GH2 igniter mixture from a blowdown system initiates combustion. Gaseous nitrogen (GN2) systems in the X-43, adapter and B-52 are used for compartment purge, line purge, and coolant pressurant. Pressure-fed water-glycol coolant systems in the research vehicle and adapter, with tanks incorporating bladder positive expulsion devices, provide cooling for the engine during high speed flight.

Objective

The goal of the X-43 fluid system development effort is to ensure their proper operation, for flight safety and mission success.

Justification

The fluid systems contain a large amount of energy, both in the form of combustible chemicals and pressure, and therefore constitute a major potential source of hazards in the flight program. Also, propulsion systems have historically been a major cause of failures for expendable launch vehicles.

Approach

First, appropriate design and testing requirements were established. Performance requirements were based on needs of the scramjet engine and its research objectives. Hardware design and testing requirements were based on guidance of established military standards, and to meet particular NASA requirements. A mindset was taken like that of expendable launch vehicles and spacecraft, where the mission is a one shot deal and the systems must work correctly the first time, with little or no human intervention.

Vehicle space limitations led to the use of higher pressures, custom designed components, and demanding performance requirements, thereby increasing risk and developmental effort. The contractor designers implemented palletized subsystems, which allowed for subsystem testing independently of the vehicle.

Component and subsystem testing were performed by contractors and vendors. Qualification testing on components determine if the design is sufficiently robust for the operating conditions. Verification testing on components and subsystems determine if they meet the performance requirements. Acceptance testing at all levels check the integrity and functionality of the hardware.

NASA will perform validation testing on the installed systems in the vehicle, to ascertain the systems can perform the mission. Captive carry flight will be the final test of system integrity prior to the scramjet test flight.

Status

Testing to date have been highly successful in troubleshooting, understanding and maturing the components and systems, as intended. Adjustments and minor changes were made to the hardware, as issues were identified by testing. Subsystem performance verification testing was indispensable for designing the system control logic. Validation tests in the vehicle are expected to commence shortly. All these tests build confidence for safe and correct operation of the fluid systems in the vehicle.

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X-43A Environmental Control Systems Performance

Summary

The Hyper-X research program, conducted jointly by NASA Dryden and NASA Langley, was conceived to demonstrate a scramjet engine in a flight environment. The X-43A Research Vehicle, the instrument of the Hyper-X program, will be lofted to its pre-determined research test condition with the aid of a modified air-launched Pegasus booster. After separation from the launch vehicle, the research vehicle will conduct engine operation tests, and then perform PID maneuvers at selected points in the deceleration phase.

One of the many facets critical to achieving mission success of the operation is control of the vehicle environment. Deflagration and detonation hazards exist due to the nature of the ignitor and fuel that are used for the SCRAM jet engine experiment. In addition to this hazard, there are thermal issues affecting guidance and navigation, instrumentation and control systems equipment, and propellant feed system performance. Therefore, effective mitigation is critical to keep both vehicle and crew safe during the mission phases leading up to drop from the B-52 and ensuring mission success.

Objective

Dryden's Propulsion efforts have, in part, focused upon mitigation of the fire hazards and assurance that the operating environment will be maintained until completion of primary mission objectives. Considerable effort has been expended on analyzing; leak detection capability and sensitivity, deflagration and detonation characteristics of fuel and ignitor, gas species detector performance, purge capability to transport leaking fuel and oxidizer, inert vehicle cavities, and cool equipment, and nitrogen system performance. Validation testing on environmental control system performance will be conducted for all flight phases.

Results

Dynamic calibrations to temperature and pressure compensated species detectors demonstrate good performance for entire flight phase. Analysis of the first purge system tests have indicated that further efforts must be made to seal the panel seams of the hull to increase pressure differential between inside and outside. This will insure that the purge operates as designed (see Figure 1) and can overcome aerodynamic ram air and maintain an inert internal environment. Environmental testing to date has indicated that the hazard of exceeding operational temperature limits due to internal equipment heat generation does not appear to be a problem, but more comprehensive tests are pending. The results from the first heat test are shown in Figure 2. The nitrogen system model indicates that the system will be depleted approximately 1 minute into its flight. This data is shown in Figure 3.

Status/Plans

Analysis of pressure differential and hull leakage continues. Further testing with ultrasonic leak detectors along panel seams will continue to identify areas of inadequate seals. Fully integrated vehicle thermal and inert validation tests are scheduled for early March of 2000. The true measure of purge effectiveness will be determined from the first captive carry flight on the B-52 with inerts in the ignitor and fuel tanks.

Contact

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Hyper-X Research Vehicle Purge Path and Fire Instrumentation

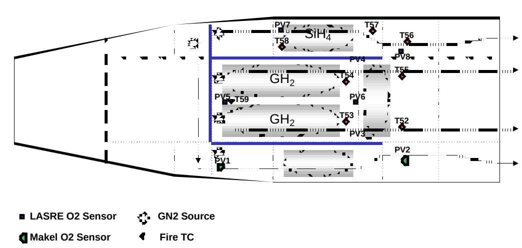

Fig. 1

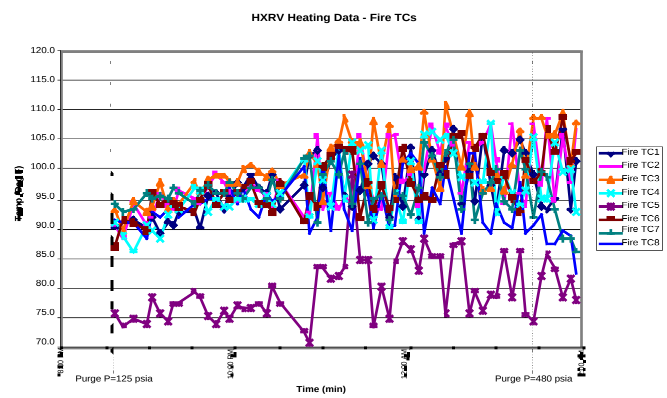

Fig. 2

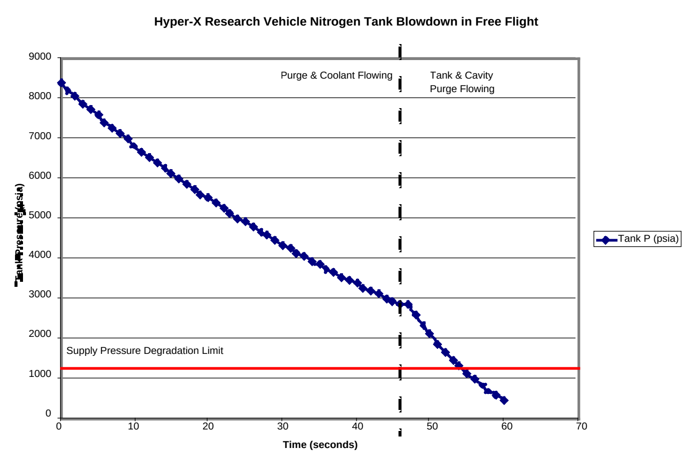

Fig. 3

Advances in Leak Detection Technologies

Summary

LASRE, X-33, Hyper-X, and X-34 are research programs meant, in part, to demonstrate advanced propulsion systems. In order to demonstrate the performance of these systems, high-energy fuels and oxidizers such as gaseous hydrogen, liquid hydrogen, and liquid oxygen are employed. These types of fuels and oxidizers are inherently hazardous, posing a deflagration, or worse, detonation threat to mission success of the programs.

One of the many facets critical to the mission success of these flight test operations is the flight safety of the vehicle. Being able to detect leakage of particular propellant species before they manifest into a hazard gives the program a greater likelihood of mission success. This requires a clear understanding of the limits detectable by species sensors and algorithms used to enhance species detectors.

Objective

Dryden's Propulsion group is analyzing the effectiveness of species sensors to detect and assess leakage, and understanding the leak rate relationship between inert and high-energy propellants when inerts are substituted into the propellant feed system. Several commercial sensors and a prototype sensor will, or are in the process of, being tested for their ability to accurately, and quickly, detect their respective species. Analysis of the data will determine if effective detection capability over a wide range of temperature and pressure is possible. If suitable, the instruments can be employed on various flight research experiments requiring this capability. Analysis of the inert and high energy propellant leak testing will help to scale leakage from gaseous propellant feed systems without the use of actual propellants.

Results

Commercial-off-the-shelf (COTS) oxygen sensors that were intended for automobile and medical use have proven quite effective for monitoring oxygenated environments. The calibrations demonstrated an accuracy of 1% at pressure altitudes up to 50 kFt. The uncertainty analysis results for these instruments are shown in Figure 1. A fully pressure and temperature compensated oxygen sensor design that was calibrated and dynamically tested, demonstrated accuracy to within 0.7% while operating at a 60 kFt/min ascent rate from 18 kFt to 105 kFt. The results of this dynamical response test are shown in Figure 2. A prototype sensor that measures speed of sound changes has had significant success in detecting hydrogen concentrations in nitrogen. The results of this sensor are shown in Figure 3.

Status/Plans

The COTS oxygen sensor and the fully compensated oxygen sensor have finished calibrations and are integrated into the Hyper-X for flight testing. Validation testing will begin sometime in mid-March. A fully compensated hydrogen detection sensor is scheduled for calibration and dynamic testing in mid-February. If testing meets requirements, it too will be integrated into the Hyper-X flight test vehicle. Negotiations are presently under way to procure another prototype speed of sound sensor for characterization and calibration by Dryden. A research grant with UCLA is in progress to determine a leak rate correlation between gaseous hydrogen and gaseous helium leaking through orifices simulating leaks.

Contact

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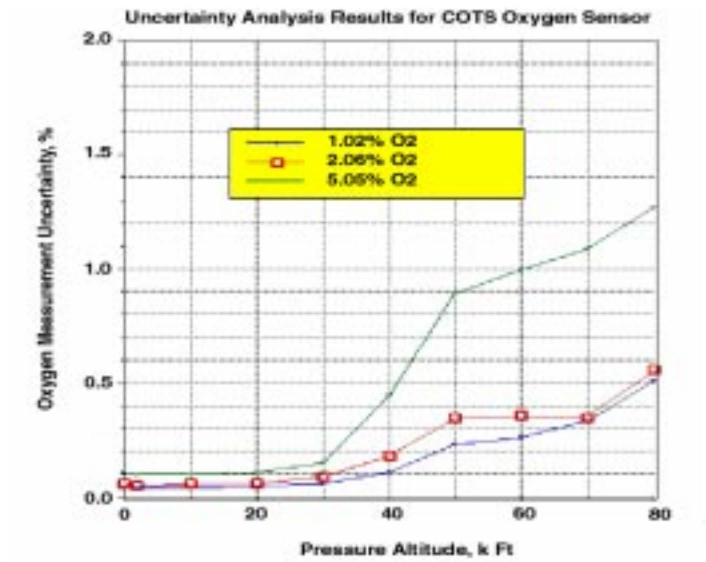


Fig. 1

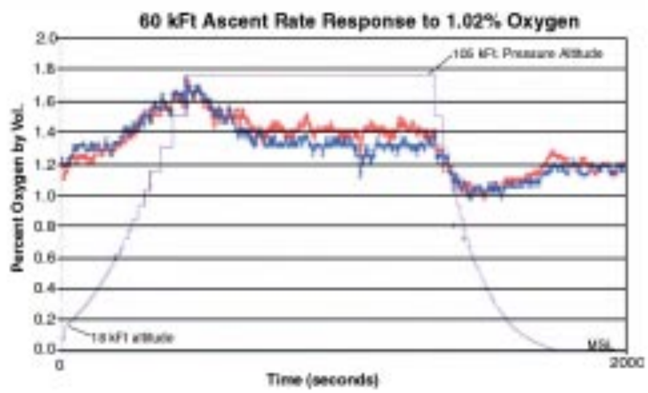


Fig. 2

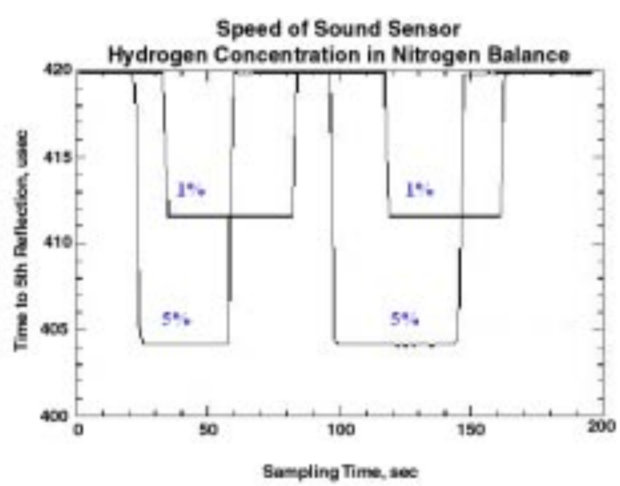


Fig. 3

Hyper-X Propulsion Flight Research Plans

Summary

The Hyper-X Flight Project, conducted jointly by NASA Dryden and NASA Langley, will develop and flight test three hypersonic vehicles that are designated X-43A. The X-43A Research Vehicles will demonstrate the performance of a scramjet engine in a flight environment. Each fully autonomous vehicle is 12 feet long, has a 5 foot wingspan, and is powered by a single airframe-integrated scramjet propulsion system. The initial X-43A Research Vehicle will be dropped from the NASA Dryden B-52 and lofted to its pre-determined research test condition (Mach 7, 100,000 feet) with the aid of a modified Pegasus booster. After separation from the Pegasus booster, the X-43A will follow a nearly ballistic flight path during the engine test phase. The sequence of events during the engine test are: 1) open inlet cowl, 2) initial engine tare, 3) engine ignition, 4) engine burn, 5) post engine tare, 6) parameter identification maneuvers and 7) close inlet cowl. The vehicle will then splash down in the Pacific Test Range with no plan to recover it.

Objectives

The overall program objectives include: (1) evaluation of the in-flight performance of an airframe integrated hydrogen fueled, dual-mode scramjet powered research vehicle at Mach 7 and 10; (2) demonstration of controlled powered and unpowered hypersonic aircraft flight; and (3) obtaining ground and flight data to validate computational methods, prediction analyses, test techniques, and operability that comprise a set of design methodologies for future hypersonic cruise and space access vehicles.

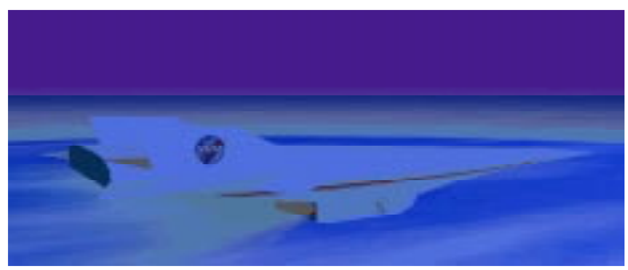
Justification

Supersonic combustion has been demonstrated since the 1950's in wind tunnel tests. However, the uncertainties in the measurements and test techniques have made actual scramjet engine performance open to debate. The X-43A will provide the first free flying scramjet engine performance data.

Approach

Because the engine is so highly integrated into the hypersonic vehicle the predicted powered and unpowered vehicle aero-propulsive forces and moments and the resulting predicted vehicle dynamics will be compared with the actual vehicle dynamics through trajectory reconstruction. A simplified first order success criteria from the overall vehicle standpoint is that the vehicle accelerate under power. It is also desirable that engine and engine component performances be determined from the flight data and compared to predictions and ground test data. Three separate but complimentary approaches are planned for post-flight data analysis. These are: 1) integrated vehicle performance through trajectory reconstruction to determine vehicle acceleration, 2) integrated vehicle performance using the engine test phase sequence to identify force increments, and 3) using pressure and temperatures to determine engine component performance.

Key to the success of each of these methods is a realistic stackup of uncertainties and a very strong effort to reduce these prior to flight.



Hyper-X Research Vehicle

Status

Monte Carlo simulation runs are being made to identify the major parameters affecting the engine test phase. Wind tunnel testing continues at NASA Langley to help identify the scramjet engine performance.

Contacts

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Development of Scramjet Skin Friction Gages

Summary

NASA Dryden is working with Virginia Polytechnic Institute and State University (Virginia Tech) to develop a skin friction gage for scramjet flight test application on the X-43. Although the Virginia Tech skin friction gage has been used extensively in scramjet wind tunnel tests, it has never been used in flight. As the results, an extensive development, testing, and validation program is being pursued to help develop the best possible gage technology for the X-43 scramjet flight application.

Objective

Develop and validate sensor technologies for skin friction measurement inside scramjet engines of hypersonic flight vehicles such as the X-43.

Justification

Surface skin friction force is one of the important forces affecting hypersonic flight vehicles and their propulsion systems. For scramjet engines such as those used in the X-43, scramjet skin friction drag is a significant portion of the net engine thrust. Accurate measurement of the skin friction force is important to validate analysis tools, provide wind tunnel to flight correlation, and determine scramjet engine efficiency.

Approach

The Virginia Tech scramjet skin friction gage uses the cantilever-beam design. In this design, the shear stress produced by the flow above displaces the sensor head at the top of the beam and produces strain at the base of the beam. Semiconductor strain gages installed at the base of the beam measure the strain. The strain gage output is proportional to the wall shear stress.

Although the gage concept described above is simple, it is quite a challenging task to build a flight gage that would work inside a scramjet combustor. The gage will have to operate in an environment of extreme pressures, temperatures, and heat transfer rates. Also, effects such as in-flight g-loading and vibration have to be considered. Currently, the following tests are being planned:

- The F-15B/FTF flight test.
- The NASA Glenn Trailblazer Mach 6.4 scramjet test at GASL.
- The Hyper-X engine model (HXEM) Mach 7 scramjet test at NASA Langley.
- The Trailblazer Mach 3 direct-connect combustor test at NASA Glenn.

Plans are being made to test skin friction gages in the X-43 flight no. 2 (Mach 7) and flight no. 3 (Mach 10).

Results

In July 1999, the scramjet skin friction gage design was tested successfully in the NASA Glenn Mach 6.4 Trailblazer scramjet test at GASL. Trailblazer was an RBCC launch vehicle concept at NASA Glenn. It used a scramjet as part of its operating cycle, and the GASL test provided an excellent opportunity to validate the X-43 gage concept in a relevant environment. Good skin friction data were obtained for three hydrogen−fueled scramjet tests. The skin friction data for one of those tests is shown in figure 2 below. The skin friction gage performed well during the GASL test with no major problems.

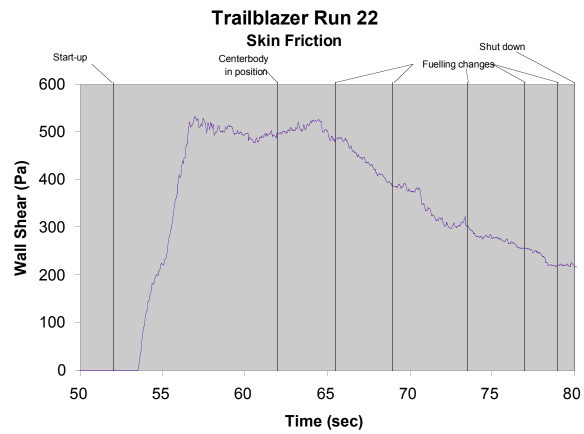

Figure 1. Trailblazer Mach 6.4 Scramjet Skin Friction Data.

Work is currently underway to build and install scramjet skin friction gages in the NASA Langley Mach 7 HXEM Arc-Heated Scramjet Test Facility (AHSTF) test. A skin friction gage prototype for the HXEM test is shown in Figure 2 below.

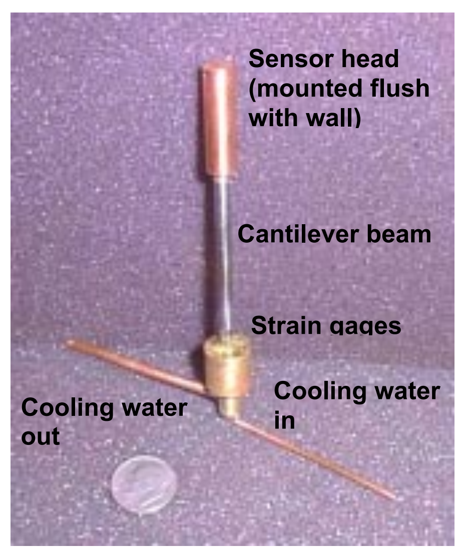

Figure 2. HXEM Skin Friction Gage Prototype.

The HXEM skin friction test is scheduled to take place in March 2000. HXEM is a wind tunnel model of the X-43 scramjet engine. This test will allow the evaluation of the skin friction gage under the realistic operating conditions of the X-43.

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Gust Monitoring and Aeroelasticity Experiment

Summary

The Gust Monitoring and Aeroelasticity (GMA) experiment was initiated by University of California, Los Angeles (UCLA) graduate students under the UCLA/NASA Center for Flight Systems Research. A small test wing was flown on the NASA Dryden F-15B Flight Test Fixture II (FTF-II) flight data on gust monitoring and aeroelasticity.

Objective

The primary purpose of the GMA flight experiment was to measure the aeroelastic response of a small test wing, and to monitor in-flight gust intensity using a customized laser-based device.

Approach

The small test wing and a low-power laser were flown attached to the aft/left panel of the FTF-II test article. The small, flexible 18-inch long, 4-inch chord, test wing was instrumented with accelerometers and strain gages to measure the dynamic response of the wing and FTF-II. Video cameras mounted on the underside of the aircraft recorded the movement of the test wing.

Results

The GMA experiment was flown on the F-15B / FTF-II at speeds of up to Mach 0.80 and altitudes up to 20,000 feet over a series of 4 flights. Flight data was used to investigate the relationship of the dynamic response of a flexible wing to in-flight maneuvers and atmospheric wind gusts. In addition, the experiment tested a new gust-monitoring device based on measurements of the forward scattering of a laser beam. New aeroelastic modeling techniques are being developed and will be tested using data collected during this experiment. These modeling techniques are being developed to aid in the synthesis of active gust alleviation and flutter suppression control algorithms.

Fig. 1. F-15B with FTF-II

Fig. 2. GMA Experiment on FTF-II

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F-15B Propulsion Flight Test Fixture (P-FTF)

Summary

Many new, experimental propulsion systems are approaching Flight Test, including Pulse Detonation Engines (PDE) and components, Rocket-Based Combined-Cycle Engines (RBCC) and components, Advanced Inlets, and Aerodynamic Models. NASA's F-15B provides a unique, low-cost opportunity to develop analysis techniques for advanced propulsion concepts: (1) Subsonic/Transonic/Low Supersonic data points, (2) Relatively simple flowpaths (PDE, RBCC), (3) Analysis tool validation, (4) Gain valuable experience in PDE / RBCC in-flight force measurement & flight test techniques prior to full-scale testing. The Propulsion Flight Test Fixture (P-FTF) is designed to support these efforts.

In the past half-year, the P-FTF has evolved from an intangible concept into a design ready for manufacture. The current design has two main external features: a pylon and a flight experiment (initially a simple "tube" for flow quality analysis). The pylon is designed for mounting the entire fixture to the F-15B centerline pylon, transfer all loads into the F-15B suspension lugs, and to house instrumentation and tankage for the propulsion experiment.

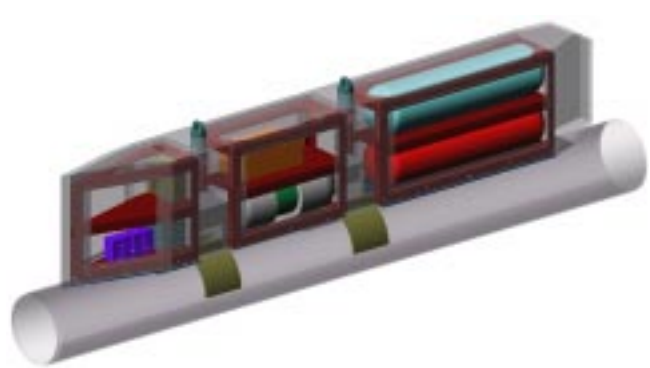
The flight experiment is attached to the pylon by a unique two-point force balance. The in-flight force balance separates the flight experiment from the pylon and measures the forces encountered or generated by the experiment while in flight. The balance is the result of a novel approach to meeting several stringent design criteria, including geometric constraints within the pylon, providing a method of mating the pylon to an experiment, tolerance of maximum forces and moments expected in-flight, catastrophic failure protection, and the versatility to accommodate future requirements that are currently unknown.

There are three main bays within the P-FTF pylon. Each bay contains a specially designed rack. The racks are easily removable from the lower cavity of the pylon, which allows for simple manipulation and setup of the experiment components. The forward bay will notionally contain the instrumentation for the experiment, while the other two bays (mid and aft) will contain additional instrumentation as needed and the main tankage: one insulated LOX tank, one N2 or He pressurant/purge tank, and one fuel tank (JP, etc.).

The P-FTF is expected to be manufactured and flying within the next year.

NASA 836 In-Flight with P-FTF Concept Mounted

The Propulsion Flight Test Fixture



Internal Configuration

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F-15 Skin Friction Flight Test

Summary

NASA Dryden is working with Virginia Polytechnic Institute and State University (Virginia Tech) to develop a skin friction gage for scramjet flight test application on the X-43. Although the Virginia Tech skin friction gage has been used extensively in scramjet wind tunnel tests, it has never been used on a flight vehicle. The F-15B/flight test fixture (FTF) provides an excellent opportunity for testing the Virginia Tech gage in the flight environment.

Objective

Evaluate and validate the Virginia Tech skin friction gage concept in flight using the F-15B/FTF.

Justification

Surface skin friction force is one of the important forces affecting hypersonic flight vehicles and their propulsion systems. For scramjet engines such as those used in the X-43, skin friction drag is a significant portion of the net engine thrust. Accurate measurement of the skin friction force is important to validate analysis tools, provide wind tunnel to flight correlation, and determine scramjet engine efficiency.

Approach

A diagram of the Virginia Tech skin friction gage concept is shown in figure 1 below.

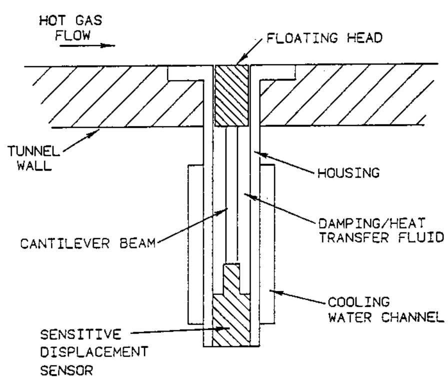

Figure 1. The Virginia Tech Skin Friction Gage Concept.

The non-intrusive floating head is mounted flush with the solid surface. The shear stress produced by the flow from above displaces the floating head slightly. This produces strain at the base of the cantilever beam. Extremely sensitive displacement sensors measure the strain at the base of the beam, and the voltage output of the sensors will indicate the actual skin friction shear stress.

In the F-15B/FTF flight experiment, the skin friction at a location on the surface of the FTF II will be measured in flight using the Virginia Tech skin friction gage, the Preston tube method, and the Clauser plot method. The flight conditions will be chosen so that the flow over the FTF II approximates the simple flat plate flow, and a good estimate of the skin friction can be obtained using the Preston tube and the Clauser plot methods. These estimates can then be used to evaluate the accuracy of the Virginia Tech gage.

Results

To date, an F-15B/FTF skin friction sensor complex has been built. This complex was designed to fit into existing 8-inch hatches on either side of the FTF, facilitating joint flight testing with other FTF experiments. Shown in figure 2 below, this skin friction sensor complex includes the boundary layer rake, RTD temperature sensors, heat flux gages, and a Preston tube. The skin friction gage will be mounted on the plate between the rake and the Preston tube.

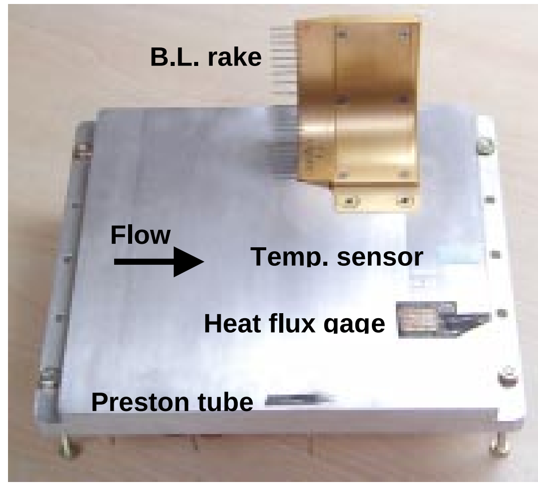

Figure 2. The F-15B/FTF Skin Friction Sensor Complex.

In March, 1999, the boundary layer rake and the Preston tube were tested in the NASA Glenn Research Center (GRC) 8 × 6 Supersonic Wind Tunnel at Mach numbers 0 to 2 with excellent results. As can be seen in figure 3, skin friction values computed from the rake and the Preston tube data agree with theory to within ±5 percent.

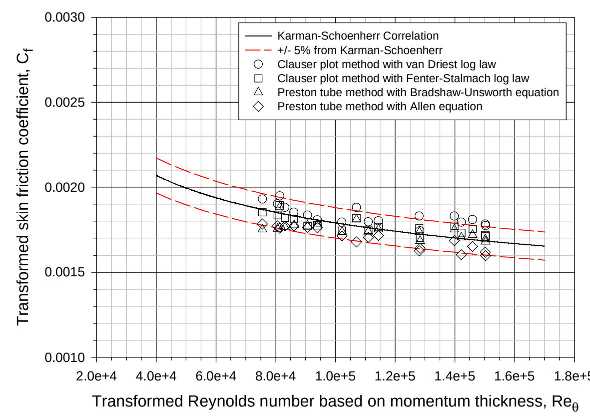

Figure 3. The Wind Tunnel Skin Friction Results.

The skin friction sensor complex (without the skin friction gage) was flown together with the F-15 Gust Monitoring & Aeroelasticity (GMA) experiment in September, 1999 . The skin friction gage is planned to be installed in the skin friction sensor complex and flown together with the F-15 Hot-Wire Anemometry experiment in February, 2000.

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X-43A Propulsion System Controls Update

Summary

The Hyper-X research program, conducted jointly by NASA Dryden and NASA Langley, was conceived to demonstrate a scramjet engine in a flight environment. The X-43A Research Vehicle, the instrument of the Hyper-X program, will be lofted to its pre-determined research test condition with the aid of a modified air-launched Pegasus booster. After separation from the launch vehicle the propulsion system will be commanded to follow a prescribed ignition and fueling sequence. Key to mission success is the ability to accurately measure and control fuel flow rates, as well as avoiding inlet unstart and flameout stability limits. Current propulsion system control (PSC) design features active stability control and fault accommodation.

The propulsion control laws rely on adequate engine performance and fuel system operability knowledge. Langley full-scale engine wind tunnel and GASL fuel system testing have proven invaluable to algorithm development. Note that the design, analysis, integration and implementation of propulsion control software are the responsibility of a joint industry and NASA team of which Dryden is only a part.

Objective

Dryden Propulsion Control efforts over the past year have focused on improving the PSC algorithm performance in a detailed design phase and planning for pre-flight validation tests of the Research Vehicle. Of prime consideration has been ensuring a fueling sequence that maximizes the likelihood for a successful scramjet engine ignition and operation. Considerable effort has been expended in gathering and reducing GASL fuel system data. The data is being utilized to verify fuel system performance with flight hardware. Fault accommodation has been developed to improve control law robustness to single point failures of critical engine and fuel system sensors.

Results

Fuel system and engine performance simulation model development is nearing completion. The mass flow estimation and valve timing sequence are being finalized for an updated release of the PSC flight software. Fuel system operation and performance has been improved by accounting for test hardware and fluid characteristics observed during GASL fuel systems performance tests. Langley wind tunnel data was used to devise and demonstrate logic that will detect and respond to the onset of inlet unstart. Current fuel system and wind tunnel results predict successful engine operation with adequate fuel supply and ample stability margins.

Status/Plans

The control system has matured considerably over the past year and final flight software expected to be delivered by mid-February of 2000. Hardware-in-the-loop and aircraft-in-the-loop tests are scheduled during the first half of CY99 where vehicle system validation tests are expected to increase confidence in hardware integration - including control laws.

X-43A Flight Research Vehicle being prepared in Dryden hangar.

Hyper-X Flight Engine Simulator in Langley 8’ hypersonic wind tunnel

Contacts

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Development of Remote Site Capability for Flight Research

Summary

Lower costs and higher productivity for flight research and flight test activities can be achieved through effective use of network communications. There are two characteristics of flight test environments that impact how effective communications should operate. First, it is recognized that data analyses and signal processing is often required to extract useful information and knowledge from the data. Effective communications over a network must thus enable signal processing to be conducted at the downstream end of the network. Second, flight projects often have a substantial number of test engineers and researchers that cannot or need not be in the control room, but still require rapid access to data. This fact suggests that communications must also enable a scalable network of remote sites. Thus, a generic remote site capability for flight research enables geographically distributed teams to rapidly assimilate and process flight test data.

As an outgrowth of efforts to automate flight flutter testing, a network caching data server has been created that manages live data streams in manner suitable for use in a remote site network infrastructure. RBNB/DataTurbine server technology has been introduced as a commercial product and received recognition in NASA's 1999 Software of the Year competition.

Dryden engineers have integrated telemetry streams as data sources this past year, developed software tools to manage multiple channel groups, designed and implemented network distribution to control room and selected Dryden remote sites, implemented prototype tools for flight flutter testing and more general data visualization, and have initiated design and deployment of remote site nodes that extend Dryden's ability to support remote sites for flight operations across the Agency.

Objective

Dryden remote site efforts over the past year have focused on getting telemetry data flowing through the RBNB, applying the baseline functionality in support of flight test operations, extending the capability to meet the demands of supporting non-Dryden remote sites, and leveraging the AeroSAPIENT project to actually move data to non-Dryden remote sites.

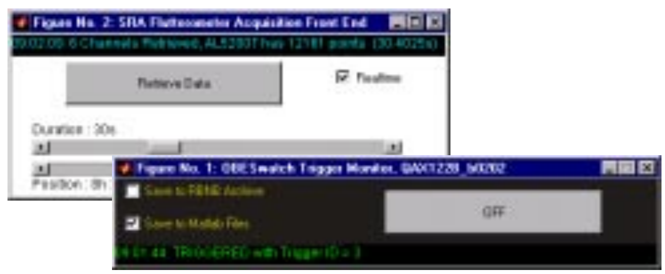
Results

A software package called REDDS (RBNB Executive Data Distribution System) establishes baseline capability to extract groups of tag/data pairs, create engineering units (EU) data and feed those channels into the RBNB/DataTurbine. Design of a second-generation version has been initiated that permits sophisticated parsing of the tag/data pairs and remote EU processing. This version splits REDDS functionality into two modules called DAPCapture and tmSplit. Connectivity to control room and remote sites in PAO and B4800 was established. Matlab-based flutterometer software for improved flutter testing diagnostics began on-line testing. A number of tools were created including:

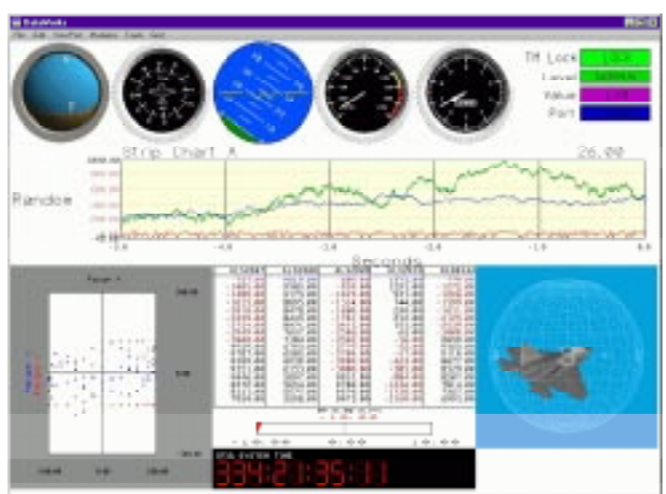
- A MATLAB-based data acquisition front-end permits commanded or data-driven triggering during flight
- A java plug-in module to RBNB enables on-demand extraction of discrete bits from packed integers
- A utility to extract time histories from RBNB and write them to disk in GETDATA and MATLAB file fomats
- An on-line version of Dryden's in-house time history tool QuickPlot was created
- An NT-based data visualization package called DataWorks provides high quality stripchart, 2D, and 3D data visualization tools

Status/Plans

Infrastructure buildup and deployment continues. AeroSAPIENT, F-18SRA/AAW and X-43 dominant users near term CY00. Initiate development of improved multi-server management and native ATM support. Validate and/or evaluate existing and in-progress security enhancements such as Entrust/PKI encryption module.



MATLAB Data Acquisition Utilities



Data Visualization Client (DataWorks)

Contacts

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Jim Harris, DFRC, FE, (661) 258-2431

AeroSAPIENT Project Initiated

Summary

NASA's Aviation Safety Program is part of a national effort to reduce the number of aircraft accidents and increase passenger safety. The Agency's role within the national effort is to accelerate relevant technology development. There does not yet exist an aviation system concept of operations for the year 2020, but all candidate concepts share fundamental requirements for the following:

- Information is available to all users in near real time
- Up to date knowledge of current state of the national airspace system, airport configurations, terminal airspace, vehicle health, etc.
- Integration of information and services such that intuitive intelligent systems exist in the air and on the ground.

These requirements dictate a comprehensive air and ground infrastructure that accommodates all types of information – including communications, navigation, surveillance, maintenance or health monitoring, etc. - in a timely manner that maintains integrity and security of the overall system. In an effort to examine the feasibility of bringing the aeronautical community into the global information network, the IT R&T Base Program has created and is sponsoring the AeroSAPIENT project (Aeronautical Satellite Assisted Process for Information Exchange through Network Technology) in support of the Aviation Safety Program. AeroSAPIENT is a collaborative project involving Ames, Glenn, Langley and Dryden Research Centers.

Objective

The Phase 1 objectives of AeroSAPIENT include establishing network connectivity between NASA's DC-8 Airborne Laboratory and the ground via commercial Ku-band geosynchronous satellite relay. Connectivity is enabled with new electronically steered phased array antennas developed under NASA's Aviation Systems Capacity Program. The wireless link will support multiple protocols and prioritization of data channels.

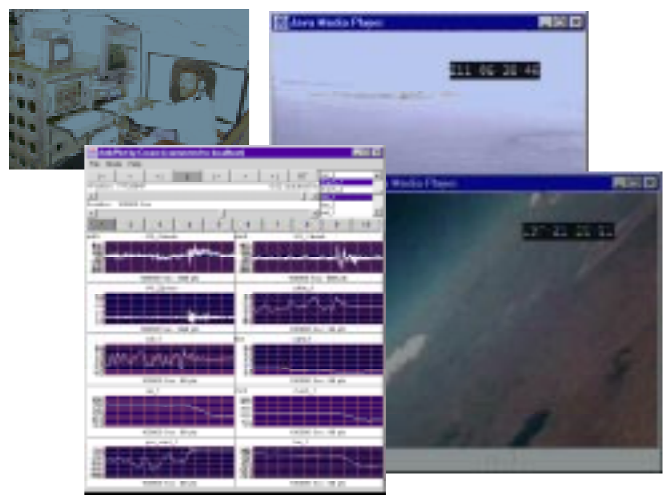
Several applications will run on board the aircraft that are representative of applications and data types relevant to aviation safety. These include

- Weather data into and out of the aircraft, including lidar-derived turbulence information (via ACCLAIM project)
- Digital voice and text messaging
- Engine monitoring and other information representative of that recorded on digital flight data recorders (DFDR).
- Link performance metrics

The aircraft will have an on board data management system based on the RBNB/DataTurbine software. When the satcom link is active, information can flow to/from the aircraft through a ground station at Glenn Research Center, where data propagates through AeroSAPIENT remote site server nodes (an RBNB/DataTurbine server network) to researchers across the Agency.

Status/Plans

A prototype on board data management node (COACT rack) was designed and evaluated as a piggyback experiment during the KWAJEX deployment July '99. Modifications to the COACT rack are in progress. Buildup of the satcom rack has been initiated. AeroSAPIENT remote site nodes have been designed and are in a build-up phase. Dryden has contributed resources and is leveraging remote site nodes for general flight test application. Most tasks for integrating applications have been defined. Plans for CY00 include deployment and checkout of remote site nodes during 1^{st} and 2^{nd} Qtr CY00. Installation of onboard systems scheduled to commence late July. Flight test over continental U.S. scheduled for August through October.



On-board Data Management Node and Data Visualization

AeroSAPIENT Remote Site Server Node

Contact

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Integration of Finite Element Analysis Predicted Strains with a Strain Gage Loads Equation Derivation Package

Summary

It is desired to create strain calibration files that are directly usable by EQDE from finite element results files. This will require development of methods, which will convert selected finite elements into strain gage and bridge outputs and arrange them into a format expected by EQDE. The equation derivation program (EQDE) main functions are calculation of strain gage influence coefficients and derivation of equation coefficients for the load equations. Demonstration of this approach will be with the F18 AAW for the analytical load calibration of aircraft structures.

Loads calibrations methods used in industry are not significantly different than those used by NASA. Currently, both NASA and industry calibrate aircraft structures by loading the actual structure with hydraulic actuators, recording the output of pre-installed strain gages, and then attempting to choose the best combination of available gages. This process is deficient in several ways. It requires engineering judgment for placement of a master group of strain gages and access to the airframe during a time, which is usually schedule critical.

Objective

The objective is to increase productivity of strain gage calibration and reduce time associated with aircraft loads calibration.

Justification

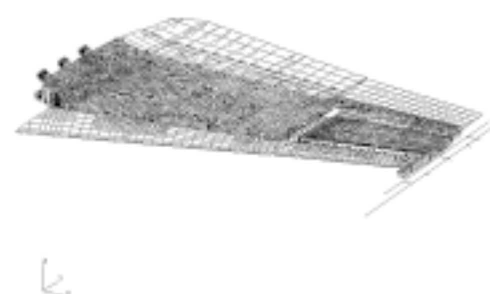
The increased cost associated with conducting load calibrations on aircraft has impacted aircraft programs. Aircraft industry is faced with economic constraints to find less expensive methods for strain gage load calibrations. Future applications could be used to develop analytical loads calibration for surface areas that cannot be loaded.

Approach

1. Create a finite element model of the structure with detail in the area of interest.
2. Identify strain gage locations to be examined by an analytical code.
3. Generate finite element strain output for analytical strain calibration.
4. Develop a code to translate analytically predicted strains by NASTRAN into EQDE
5. Use the NASA developed equation derivation (EQDE) with search algorithm to locate minimum number of gages and determine error and standard deviation for shear, bending and torque.
6. Through applied loads in NASTRAN validate calibration

Future work

This analytical procedure will then be validated against a calibration performed on the F-18 AAW through a full-scale test in the Flight Loads Laboratory. Comparison of the resulting loads equations will be made with benefits from the analytical method.



F18 Active Aeroelastic Wing

Contact

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Non-Invasive In-Flight Structural Deflection Measurement

Summary

As air vehicle design and predictive tools have matured and increased in precision there has been increased interest in not only predicting the airframe elastic shape under various flight conditions but also measuring it as well. Early in-flight deflection measurement efforts involved the use of photographic film as the sensing medium. Later efforts have involved the use of single column photo diode array sensors and infrared light emitting diode target markers. Recent improvements in the resolution and sampling rates of two-dimensional video sensors combined with new software techniques and micro computer developments have provided the potential for a new approach to this measurement task. Video motion analysis has already found application in athletic sports training feedback. Among other features, this technology potentially does away with the need for wired active targets. For this and other reasons its application to airborne structural measurement is desirable.

Objective

To begin to mature this new technology for flight use, a considerable amount of laboratory work must be accomplished first. The following objectives are planned:

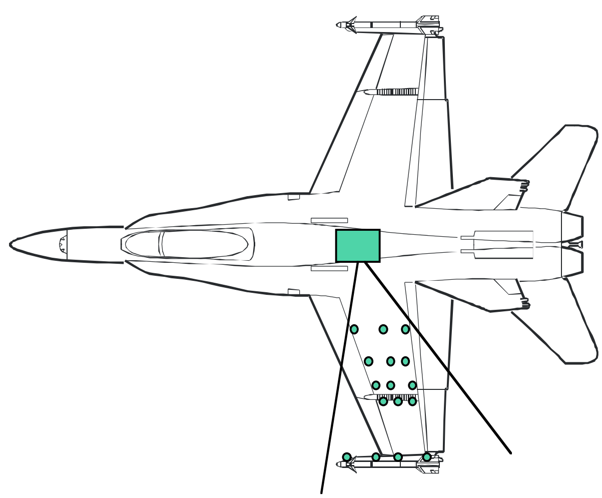
- Demonstrate a ten-fold increase in target count (relative to current Flight Deflection Measurement System, FDMS).
- Demonstrate a two and a half-fold increase in sampling rate (relative to current FDMS).
- Demonstrate a two-fold increase in measurement resolution (relative to current FDMS)
- Characterize data stream handling requirements for typical flight application.
- Identify required hardening/ruggedizing of hardware for flight environment

Approach

- Procure required hardware and software including a high speed video camera, on-board computer and storage unit, data reduction computer, and data reduction software.
- Build up a functioning ground test motion analysis system.
- Perform measurement tests with typical target ranges.
- Generate and record representative data stream.
- Supply sample data recording to Test Instrumentation Engineers to begin developing on-board data recording and TM techniques.
- Document results.

Future Work

- The longer term intention is to fly a video motion analysis system on the Active Aeroelastic Wing F-18 aircraft side-by-side with our current electro-optical FDMS and compare results. Following this, expansion to three-dimensional measurement capability could be pursued.



Active Aeroelastic Wing F-18 With FDMS

Contact

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Aerostructures Test Wing

Background/Objectives

The AeroStructures Test Wing (ASTW) will provide a functional flight test planform to validate stability prediction algorithms and flight test techniques. The next generation tools for robust stability predictions will reduce the currently high levels of risk and cost associated with flight flutter testing. The ASTW will be flown to the point of instability while real-time stability prediction algorithms will be estimating the flutter margins. The design flutter point is 0.80 Mach at 10,000 feet. The primary objective of this project is to investigate pre-flight and on-line analysis tools for load predictions and stability estimation. In-flight flutter estimation algorithms will be evaluated to determine their accuracy in safely predicting the onset of flutter instabilities and applicability towards an on-line flutterometer concept. A secondary objective is to investigate open and closed-loop control strategies for loads alleviation and flutter suppression.

Approach

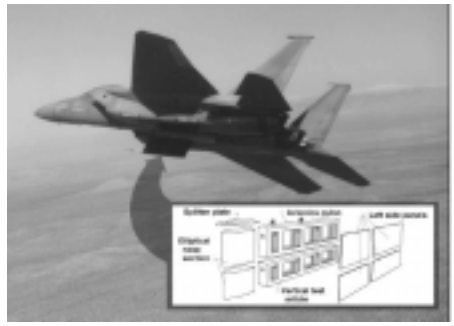
The research approach of the ASTW consists of four main components: analytical modeling, ground tests, flight tests, and technology transfer. The analytical approach will consist of using STARS and ASTROS to create structural models and predict flight loads and flutter speeds. STARS analysis will also provide a state space model required for the μ flutter prediction algorithms. The ground tests will consist of a load test to measure the static stiffness of the test wing. A ground vibration test will be done to obtain the structural frequencies and mode shapes. The ground tests will be used to update the models and stability estimations. The flight test of the ASTW will take place on the F-15B/FTF-II test bed. The main objective of the flight test will be to gather static and dynamic load data for load predictions and stability estimations. A build up approach from 25% to 99% of the flutter speed will be completed on the ASTW.

Results

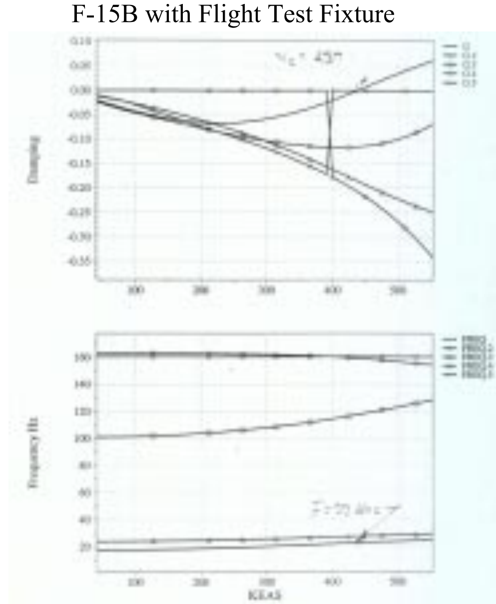
A design was completed and the wing was manufactured this last year. A finite element analysis of the completed design was completed. The first structural mode for wing bending was found to be 18 Hz and for wing torsion the frequency was 23.6 Hz. A flutter analysis of the wing estimated the flutter speed to be 437 Keas at Mach 0.80 with a frequency of 22.4 Hz.

Status/Plans

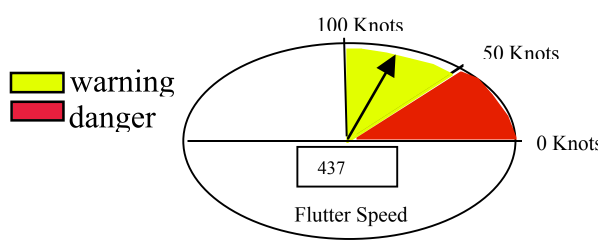
The wing is currently being mounted to the flight test fixture panel. The panel with the test wing will undergo a series of ground tests. The ground tests include a wing stiffness and strain gage calibration, a ground vibration test, and a test of the excitation system. The results of the tests will be used to update the finite element analysis and flutter predictions. The Aerostructures Test Wing should complete its flight testing during this next fiscal year.



F-15B with Flight Test Fixture



Aerostructures Test Wing Flutter Analysis Results



Flutterometer Concept, μ-Method

Contact

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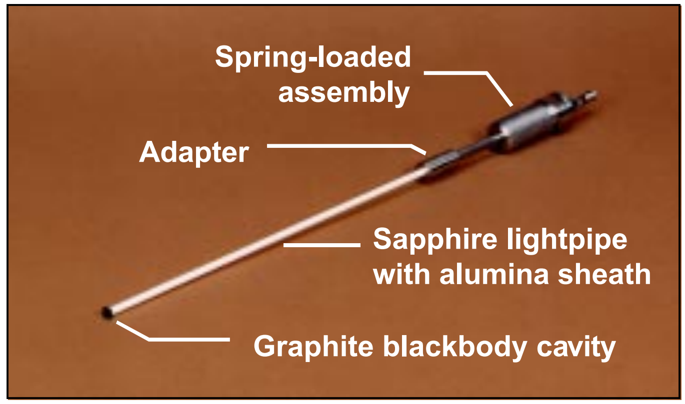
Blackbody Lightpipe Temperature Sensor Development

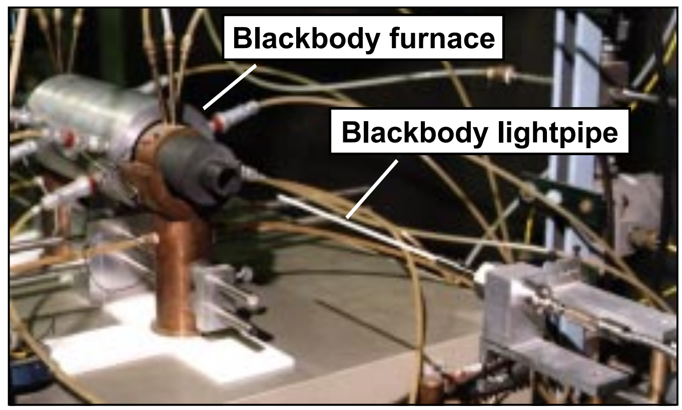
Summary

A contact temperature measurement technique using a blackbody lightpipe temperature sensor is being developed to control surface temperatures on a carbon-carbon control surface above 2000° F. A blackbody lightpipe sensor completed verification tests in the FLL blackbody calibration furnace at a peak temperature of 2732° F. The sensor was calibrated to within 2° F of a NIST calibrated optical pyrometer over a temperature range of 1472° F to 2732° F. The sensor is awaiting validation testing in an actual test application.



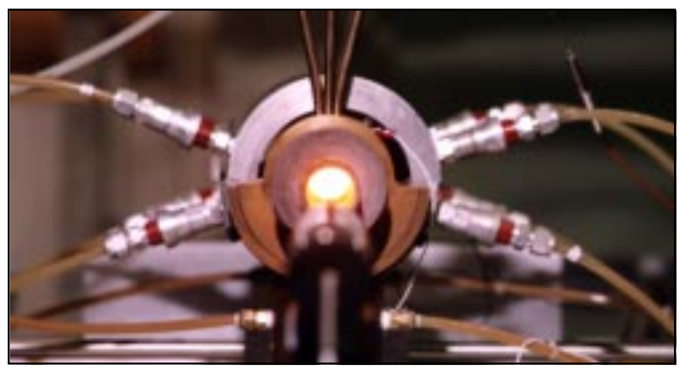
Objectives

- Develop a temperature measurement technique to control temperatures on a carbon-carbon control surface above 2000° F.
- Develop a technique to calibrate a blackbody lightpipe temperature sensor up to 2732° F.
- Perform V&V tests of a blackbody lightpipe temperature sensor.
- Calibrate the blackbody lightpipe temperature sensors required to support the Carbon-Carbon Control Surface test program



Benefit

- Establishes a method of controlling temperatures accurately and repeatedly on carbon-carbon structures at high temperatures without disturbing the substrate with bonded thermocouples



Contacts

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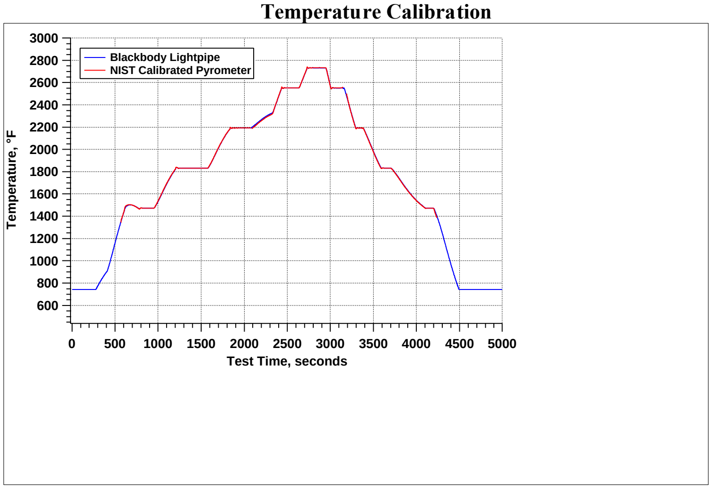
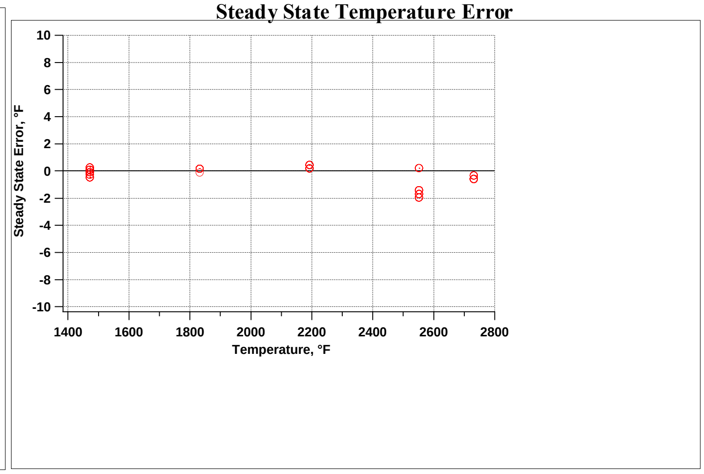
Temperature Calibration



Steady State Temperature Error



Spaceliner 100 Carbon-Carbon Control Surface Test Program

Summary

Work continued in the Flight Loads Laboratory (FLL) to establish a 3000° F test capability in support of the Spaceliner 100 Carbon-Carbon Control Surface (CCCS) test program. The three main development areas include a large aluminum test chamber, a heating system, and a water and gas cooling system. Construction of all major test systems is complete.

System checkout and integration tests are underway. The primary focus to date has been on the purge/gas-cooling system. This system utilizes a high output blower to both purge the chamber and provide cooling to the radiant heaters. The blower delivers approximately 4000 SCFM of gas at 14 to 15 psig. Liquid nitrogen is sprayed into the flow to cool the gas, which has been heated by the compression process, and displace the oxygen in the chamber atmosphere.

Checkout testing of the purge/gas-cooling system has been completed and chamber purge levels of 140 ppm oxygen has been demonstrated. Gas temperature entering the radiant heaters has been maintained at or below 70° F.

Objectives

- Establish a 3000° F nitrogen atmosphere test capability.
- Perform thermal/mechanical test up to 3000° F.
- Support the Spaceliner 100 carbon-carbon development work through test and analysis of the CCCS.

Benefits

- Provides valuable test data to the technical community on a flight-weight carbon-carbon control surface.
- Establishes methods of testing oxidation sensitive structures at high temperatures.
- Serves as a testbed to test innovative sensors.

Contacts

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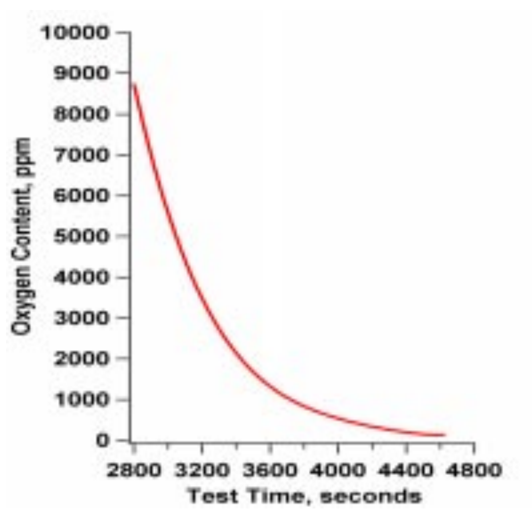
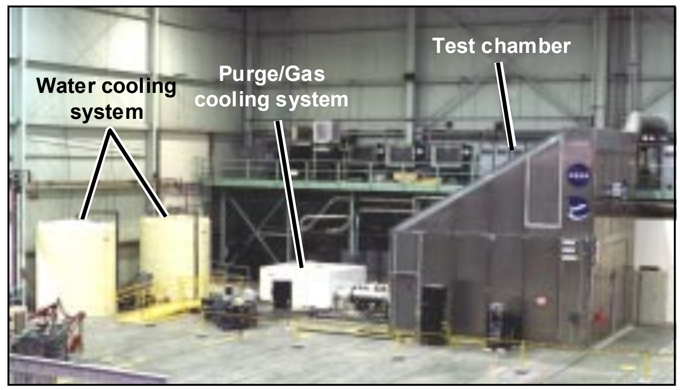
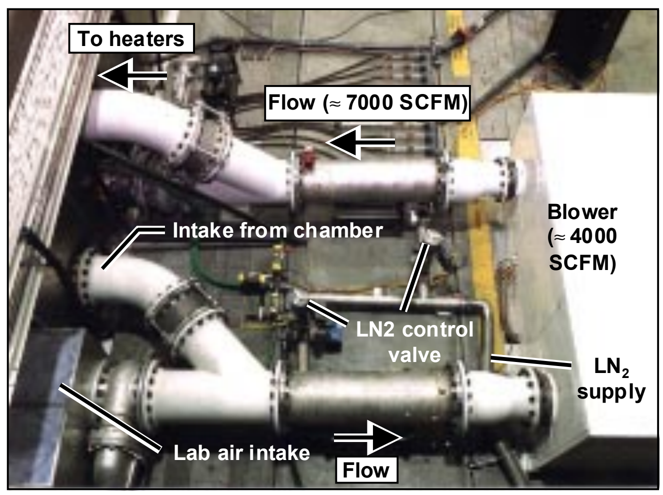
Dr. Wayne Sawyer, LaRC, (757) 864-5432




Carbon-Carbon Control Surface






Heating System

Radiant Heat Flux Gage Calibration System Characterization

Summary

Heat flux measurements typically have large uncertainties (typically 10% to 20%) associated with them. Heat flux calibration facilities being developed at the National Institute of Standards and Technology (NIST) operate at heat fluxes well below (<20 Btu/ft^2/sec) the levels achieved during high speed flight. This leaves the heat flux gage user interested in these higher heat flux levels only two options: 1) take the gage manufacturers calibration on faith or 2) develop and understand your own calibration process.

An effort is underway at NASA Dryden's Flight Loads Laboratory (FLL) to reduce the uncertainty of heat flux measurements taken at Dryden. The first phase of this effort is to thoroughly characterize the radiant heat flux gage calibration system located in the FLL. Future phases of this project will develop methods for using gages calibrated in this system in radiant heating tests performed in the FLL and flight.

Objectives:

- Characterize the radiant heat flux gage calibration system (both blackbody and flat graphite plate heaters) in the Flight Loads Laboratory in order to quantify and reduce, if possible, calibration uncertainty.
- Be able to demonstrate to customers that we understand our heat flux gage calibrations.

Challenges:

- Quantify errors associated with calibrating a water cooled heat flux gage inside a cylindrical blackbody
- Determine effect of graphite plate erosion
- Determine effect of natural convection around flat graphite plate and gage
- Determine effect of distance between flat graphite plate and gage on absorbed heat flux distribution

Recent Results:

- Steady state numerical thermal models of the blackbody have been developed and tuned to experimental data @ 2012° F (see figures at right)
- Analysis of data from the steady state numerical models and experiments show:
  - negligible convection in the blackbody
  - effective emissivity of the cylinder end wall is approx. 0.98 for insulated and uninsulated configurations

Future Work:

- Develop 3-D model of the flat plate heater for use in evaluating calibration configurations
- Develop transient numerical models of blackbody/heat flux gage calibration process
- Determine experimental heat flux using different principals of physics.

Contact:

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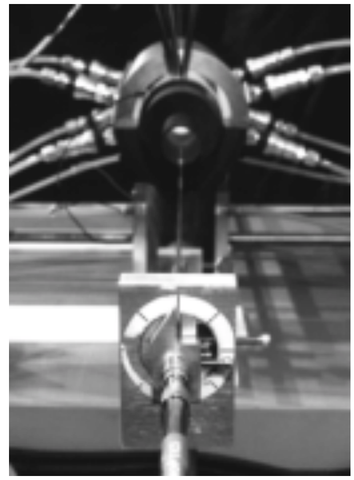
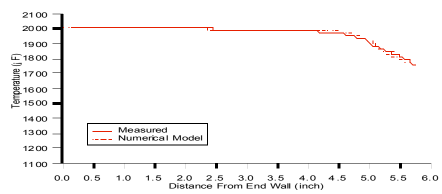
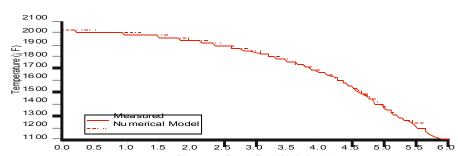


Blackbody Source with Wall Temperature Measurement Probe



Axial Temperature Distribution, Uninsulated Blackbody



Axial Temperature Distribution, Insulated Blackbody

Reference:

Jiang, S., Horn, T. J., and Dhir, V. K., 1998, "Numerical Analysis of a Radiant Heat Flux Calibration System," HTD-Vol. 361, *Proceedings of the ASME Heat Transfer Division, V. 5*, pp. 609 – 616.

Thermostructural Analysis of Superalloy Thermal Protection System (TPS)

Summary

Superalloy honeycomb thermal protection system (TPS) is a new candidate TPS for the future space transportation system. The light weight TPS must provide high temperature gradient across its core depth, stiff enough to resist bulging-out thermal bending, and its cell walls must be stiff enough to withstand buckling under external air pressure. Therefore, thermo-structural analysis of TPS is needed to obtain basic knowledge for the design of low-density honeycomb TPS panels with high heat-shielding performance, and ample stiffness to resist thermal bending and cell-wall buckling.

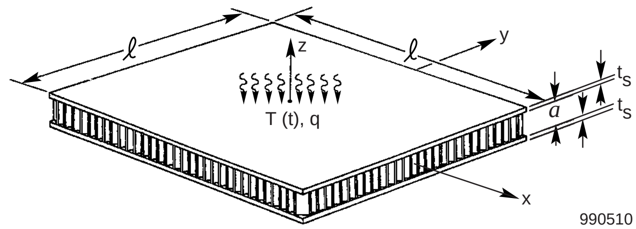


Figure 1. Honeycomb thermal protection system (TPS) under surface heating and pressure loading.

Objective

To fully understand the heat-shielding performance and thermostructural characteristics of superalloy TPS panels with varying honeycomb cell geometry, and to generate a set of theoretical curves for the designers to determine the most efficient superalloy honeycomb TPS panel structures.

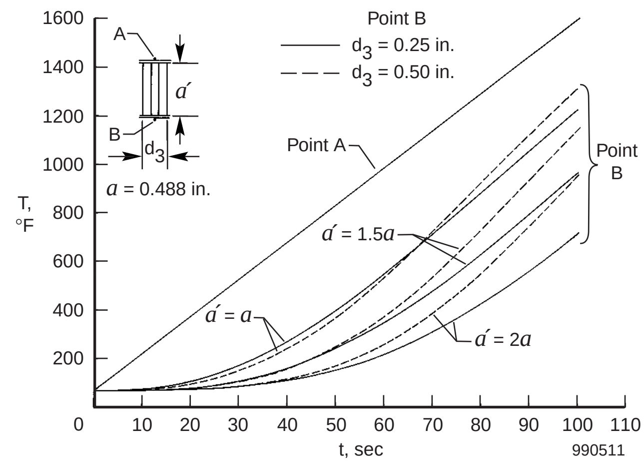


Figure 2. Effect of cell size and core depth on the heat shielding performance or superalloy TPS.

Approach

For a given superalloy, the heat shielding and thermostructural performance of the honeycomb TPS panels vary with the thickness of the face sheets, honeycomb-core depth, honeycomb cell wall thickness, and the size and shape of the honeycomb cell, etc. Therefore, thermostructural analysis of honeycomb TPS must cover the following key domains: (1) Geometrical analysis of different types of candidate honeycomb cells with the same effective density but different cross sectional shapes. (2) Heat transfer analysis of TPS with different honeycomb cell geometry. (3) Thermal bending analysis of the TPS panels subjected to one-sided heating and under different support conditions. (4) Comparative buckling analysis of different types of honeycomb cell walls under axial compression.

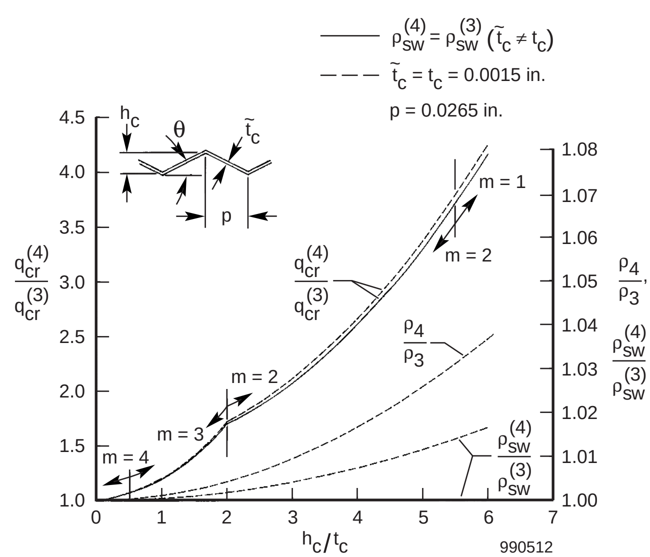


Figure 3. Plots of buckling pressure and core density of TPS cell as functions of corrugation depth.

Results

1. The heat shielding performance of honeycomb TPS panel is very sensitive to the change of honeycomb core depth, but insensitive to the change of honeycomb cell cross sectional shape.
2. The thermal deformations and thermal stresses in the TPS panel are very sensitive to the edge support conditions.
3. A slight corrugation of the honeycomb cell walls can greatly enhance the buckling strength of the honeycomb cell walls.

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Validation of the Baseline F-18 Loads Model Using F-18/SRA Parameter Identification Flight Data

Summary

The F-18 baseline loads model validation was performed as part of the Active Aeroelastic Wing (AAW) Risk Reduction Experiment on the SRA aircraft. The F-18 loads model was developed by Boeing for use with the flight simulator to determine loads at discrete locations on the aircraft. It was then modified to reflect the increased flexibility of the AAW aircraft so that it could be used in the development of the AAW control laws which are being optimized to maximize roll performance while maintaining load limits. Currently, the loads models for both the AAW aircraft and the baseline F-18 are predicting higher than expected loads especially for the Outboard Leading Edge Flap which has become the limiting factor in the AAW control law development. In order to ensure the success of the AAW program, a reasonably accurate loads model must be developed. The purpose of this experiment was to determine the level of conservatism inherent in the baseline loads model.

Approach

Three control surface hinge moments were used in this experiment: the right outboard leading edge flap, the left aileron, and the left stabilator. These measured control surface hinge moments were monitored in flight along with loads model predicted values. The loads model values were obtained by incorporating the loads model code and database into the real time fortran and using the aircraft measured flight data including: Mach, altitude, and control surface deflections as inputs to the model.

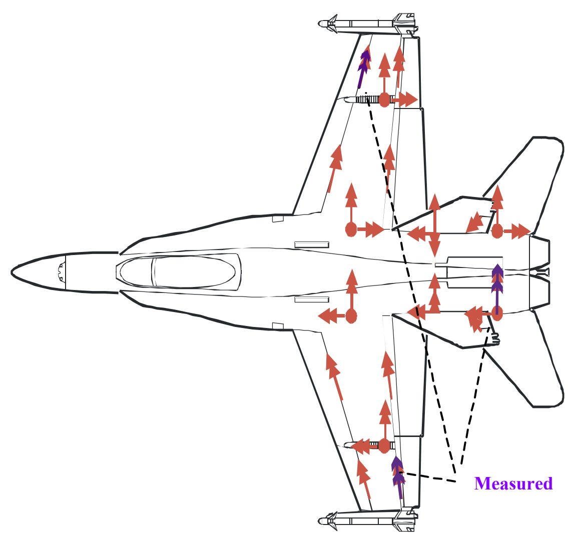
Longitudinal and lateral directional doublets were performed during which each individual control surface was deflected. Measured hinge moments were compared to loads model predicted hinge moments at twenty flight conditions ranging from Mach .85 to 1.3 and altitude from 5000 to 25,000 feet.

Results

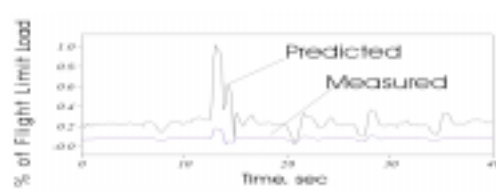
Flight test has been completed on the F-18 SRA. The first stage of data analysis has been completed in which measured hinge moments and predicted hinge moments at each test point were compared directly. The measured leading edge flap hinge moment was approximately 75% of the loads model predicted value as seen in the graph. The aileron hinge moment measured approximately 40% of the predicted hinge moment while the stabilator hinge moment exceeded the predicted by approximately 10% of the flight limit load.

Future Plans

Currently, analysis is being performed in which a linear regression technique is being used to determine the loads' derivatives for each of the hinge moments, e.g. the load on the leading edge flap due to control surface deflection. These values will be directly compared to the derivatives in the loads model database at each flight condition to determine the existence of any trends, which can be used to evaluate the database.



F-18 Component Loads Diagram
(All Loads Predicted by Loads Model)



Measured and Predicted Leading Edge Hinge Moments vs Time

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PC Based Thermal Control System

Summary
Thermal testing in the Flight Loads Laboratory (FLL) has formerly been accomplished with the use of a mini-computer/array processor combination. The FLL's third generation data acquisition and control system (DACS III) thermal control system now under development, is being designed based on a Sparc10 processor. The DACS III thermal control system is at least a year from completion. There exists an immediate need for a small channel count thermal control system that can be implemented using personal computers.

Objectives - Develop a PC based control system with the following capabilities

Use present FLL power controller cabinets for temperature control functions.

- Follow a flight profile with a rise rate ≈5º F/sec
- Provides limit checking per control channel that produces warnings and automatic shutdown when fatal limits are reached
- Permits maximum power level protection (ability to limit maximum obtainable power level for a given command)
- Power level recording
- Control switching with bumpless transfer from power level control to closed loop temperature control
- Ability to slave control zones
- Display and record all data
- View data in the control room as well as at the test site

Challenges

- Compensating for time lags between the data acquiring computer and the control computer such that the control computer is processing the correct data set
- Developing a self-tuning algorithm that automatically adjusts as it encounters various temperature ranges, control rates and material thickness

Recent Results

- Demonstrated the ability to fire heat lamps on command using the FLL's present controller cabinet receiving signals from the PC based system
- Demonstrated the ability to track a user generated thermal profile

Future Work

- Implement the remaining system requirements
- Investigate using a 'model free' adaptive control algorithm for this application

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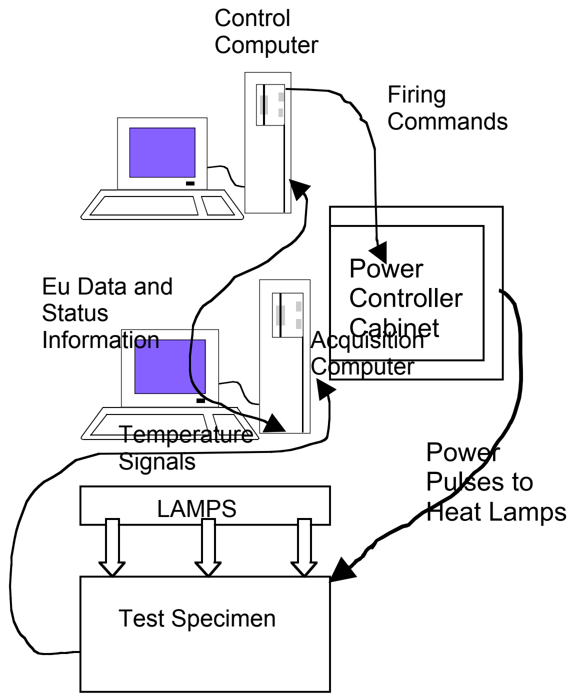
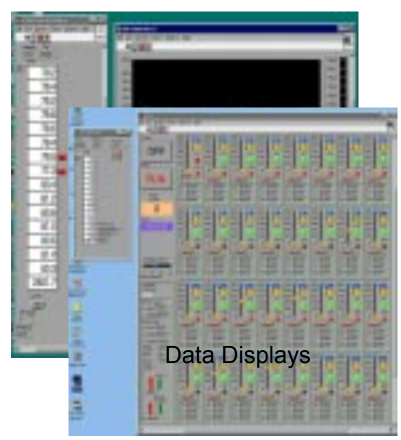




The control system consists of two personal computers linked via Ethernet. The acquisition computer receives data from the test article and converts this data to engineering units. The converted data are transmitted to the control computer. The control computer produces firing commands for the power controller cabinets based on the desired temperature profile.
The control and display features are implemented using National Instruments Labview software.

Robust Control Design for the F/A-18 Active Aeroelastic Wing

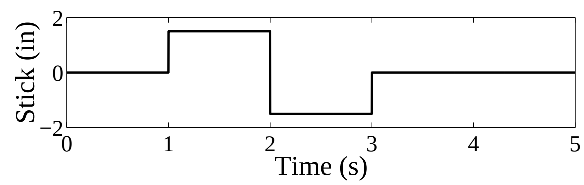
Summary:
The Active Aeroelastic Wing (AAW) is being developed as a research testbed to demonstrate technologies related to aeroservoelastic effects such as wing twist and load minimization. Currently, the F/A-18 uses a strong structure with large stabilators to perform rolling maneuvers at high dynamic pressures. The AAW will demonstrate that wing twist can be used as an efficient tool for maneuvering and that the induced loads can be reduced beyond current levels. A robust control design is formulated for the AAW using a multi-objective approach that considers all the closed-loop goals. Simulations indicate the controller simultaneously achieves maneuvering and load performance and is robust to small levels of modeling uncertainty.

Objective:
The objective of the robust control design is to synthesize a flight controller that enables the AAW to efficiently demonstrate its key technologies. In particular, the controller must induce wing twist to generate rolling maneuvers. Another objective is to minimize the loads throughout the structure during these twisting maneuvers. Furthermore, these objectives must be met without violating constraints associated with actuate position and rate limiting that may severely restrict the closed-loop performance.

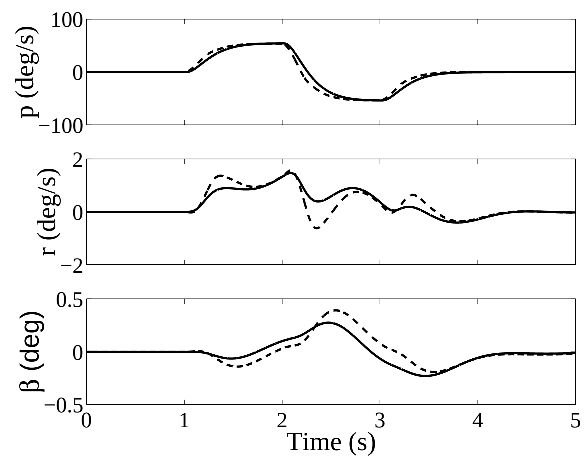
Justification:
The AAW presents an inherently multi-objective design problem with equally important criteria such as maximizing roll performance while minimizing the load on the structure. The design is further complicated by varying levels of fidelity in elements of the analytical models. Traditional and optimal control strategies are unable to efficiently consider this type of problem whereas a robust design is inherently well-suited for the AAW application.

Approach:
The controller synthesis uses a model that describes the multiple closed-loop objectives. The roll performance and handling qualities are included by formulating an error signal that describes the difference between the responses of the AAW and a current F/A-18. The load minimization is included by formulating error signals that relate the difference between the AAW loads and a set of desired loads. Also, a simple uncertainty description is associated with the model to reflect potential errors in the structural and loads models. The controller is designed using a worst-case approach that attempts to trade-off objective constraints and result in reasonable performance for both maneuvering and loads characteristics for the uncertain model.

Results:
Initial controllers have been designed for the AAW that demonstrate the concept is feasible. Linear simulations indicate the design objectives are satisfied for maneuvering and loads. Also, the controller can be realized by a simple architecture that enables efficient implementation for the flight vehicle.



Stick Command to Generate Simulated Responses



Sensor Measurements during Simulated Responses : AAW (—), standard F/A-18 (– – –)

Benefits:
- Robust control design provides multi-objective approach to achieve several closed-loop goals
- Load minimization may result in weight savings for future aircraft designs by lightening structures
- Roll maneuvering using wing twist may reduce need for inefficient and high-maintainence control surfaces

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Application of Gain-Scheduled, Multivariable Control Techniques to the F/A-18 PSFCC

Summary

The current F/A-18 flight controller was developed from a collection of independent linear controllers created at specific flight conditions. These linear controllers were combined, through gain scheduling techniques, to create a full envelope controller. Traditional gain-scheduling methods, like those employed on the F/A-18, are inherently ad hoc and the resulting scheduled controller provides no stability or performance guarantee for rapid changes in the scheduling variables. The lack of a systematic approach to gain scheduling and an absence of stability and performance guarantees are among the main motivators for current research into multivariable gain-scheduled control techniques.

The research area attempting to address the issues listed above has become known as control of linear, parameter varying (LPV) systems. The LPV methodology provides for a systematic approach that explicitly designs a global controller for all values of the scheduling parameters throughout the intended operational envelope. The resultant controller guarantees stability and performance in the presence of time-varying parameters. In addition, these controllers are robust to errors in model estimation and sensor noise while simultaneously achieving desired handling quality characteristics throughout the flight envelope.

Objective

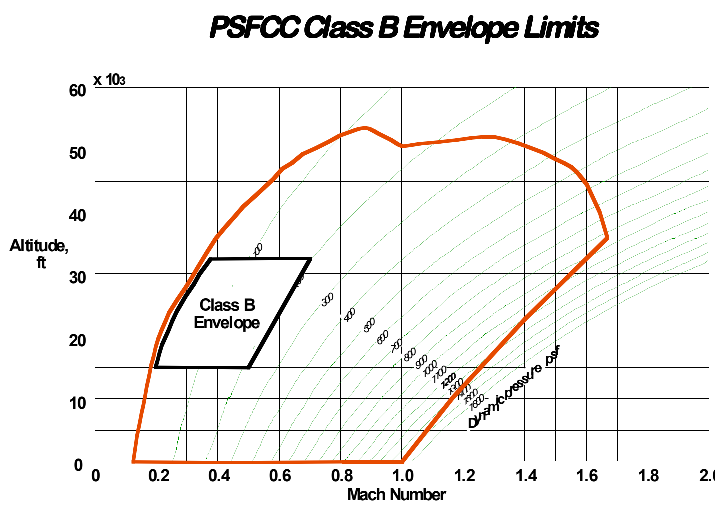
The objectives of this research are a demonstration and evaluation of the design methodology, its exposure and suitability to a flight environment, and its viability as a design approach for future applications. To achieve these objectives, the controller will be implemented on the F/A-18 PSFCC. After the controller is verified and validated in a hardware environment it will be flight-tested within a limited, low-energy, relatively safe region of the F/A-18 envelope.

Approach

Controller point designs are created within the PSFCC Class-B envelope using robust control techniques. By virtue of the robust design, all point controllers are linear, time-invariant (LTI) and are represented as state space systems. LPV techniques proceed to combine all independent point designs to create a single, composite controller valid over the entire design envelope. The resultant composite controller, however, loses the LTI nature of the point designs since the elements of its state space representation are now scheduled, through LPV techniques, as a function of flight condition. Fortunately, the LPV generated schedules can be implemented in a similar manner to traditional methods tending to reduce concerns of excessive memory and computational resource utilization.

Status/Plans

LPV designs for both sets of aircraft axes (longitudinal and lateral/directional) are complete and have been implemented in a Mathworks Simulink environment as well as in a Dryden F/A-18, six degree of freedom simulation. Piloted simulation evaluations are expected during the summer of 2000 and a flight test is tentatively scheduled for fall of 2000.



Contacts

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Fiber Optic Instrumentation Development

Summary

Fiber optic sensor technology has been the subject of considerable research interest over the past few years. This emerging technology potentially offers numerous advantages over conventional aircraft sensors. A comprehensive research effort has been underway at Dryden to develop fiber optic systems, evaluate sensor attachment performance, and evaluate this new technology in the laboratory, and through ground- and flight-testing.

Objective

Systems Development - Develop a self-contained, portable fiber optic measurement system for in-flight and ground base testing. **Sensor Evaluation -** Develop attachment techniques and evaluate the accuracy of fiber optic sensors in controlled laboratory conditions. **Ground test evaluation -** Evaluate the feasibility of using fiber optic sensors in large-scale ground-based test applications. **Flight test evaluation -** Demonstrate in-flight fiber optic sensor measurements on simple flight structures made of well-characterized materials.

Justification

A comprehensive fiber optic research program involving laboratory and flight research must be performed first to successfully implement fiber optic technology at DFRC.

Approach

Acquire off-the-self measurement systems and sensors for evaluation. Modify and develop fiber optic digital signal processing hardware and sensor attachment techniques. Evaluate sensors and systems in both laboratory and flight environments.

Results

A comparison between four fiber optic strain sensors and conventional strain gages showed agreement to within 2% after sensors were calibrated. The sensors were attached to a cantilever beam and subjected to a concentrated load at the beam tip. A vacuum/inert atmosphere furnace was also purchased to perform high-temperature evaluation of fiber optic strain sensors on a variety of material systems. The furnace completed qualification testing at 3000° F. In addition, a flight test fixture was designed and fabricated and instrumented with both fiber optic and conventional foil strain gages. The fixture was as a flight test experiment on the F-18 SRA. Test data from the flight experiment are currently being analyzed.

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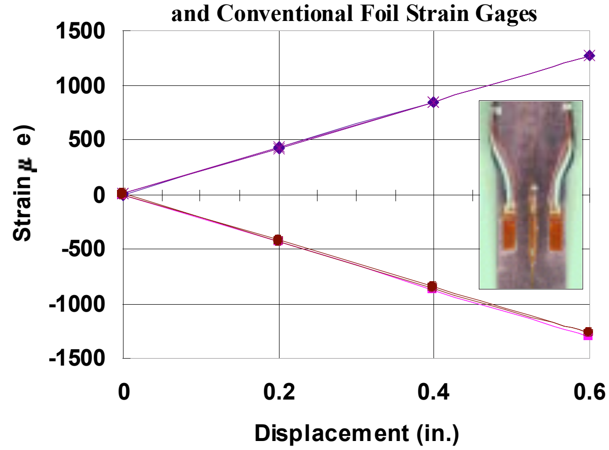
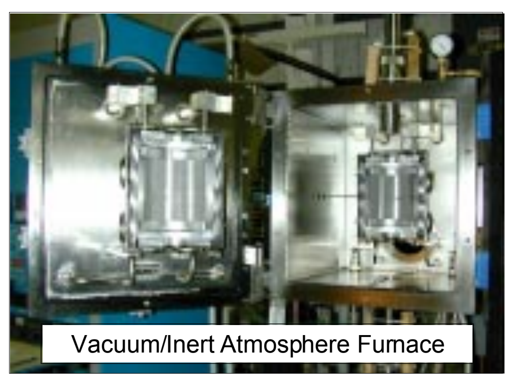
Overview of Fiber Optic Research Program

Comparison Between Fiber Optic and Conventional Foil Strain Gages





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Sangmon Lee, Linda Kelly, Art Lavoie, Kiefer, Champaigne
RF Wireless Flight Control System

Al Stewart, Allen Parker, Knut Roepel, Van Tran, Manny Castro, Ron Rough, Chris Talley
Third Generation Data Acquisition Now in Operation
Published October 1999

Dave Richwine, Craig Stephens, Kirsten Carpenter, Michelle Greslik, David McAllister
Durability Studies on the F-15B Flight Test Fixture
Published July 1999

Alex Sim, Jim Murray, Tony Frackowiak
Model X-33 Subscale Flight Research Program
Published October 1999

Klaasen (SVT Assoc/SBIR company)
UV Dual and Photodiode Sensors for Dynamic Combustion Control

Patents

Brent Cobleigh, Tony Whitmore, Ed Haering
Flush Airdata Sensing (FADS) System Calibration Procedures and Results for Blunt Forebodies

Bryan Duke
Algorithm and Test Technique for a Frequency Excitation Using an Optimized Multiple Frequency Sweep Waveform (Sweepstack)

Paul Curto, Gerry Brown, and Jan Zysko (NASA HQ)
Device for Providing AC Cockpit Display of Runway Status, Local Wind Velocity & Direction, and Global Position

Mark Nunnelee
Force Measuring C-Clamp

Anthony Piazza
Alumina Encapsulated Strain Gauge, Not Mechanically Attached To Substrate, Used to Temp Compensate an Active High Temp Gauge in Half-Bridge Configurations

Space Act Awards

David Hedgley
A Formal Algorithm for Routing Traces on A Printed Circuit Board

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Selected research and technology activities at Dryden Flight Research Center are summarized. These activities exemplify the Center's varied and productive research efforts.

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